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APOLLO LUNAR DESCENT AND ASCENT TRAJECTORIES

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ABSTRACT

A description of the premission planning, real-time situation, and postflight analysis for the lunar descent and ascent phases of the Apollo 11 mission, the first manned lunar landing, is given. Actual flight results are shown to be in agreement with premission planning. Based on Apollo 11 postflight analysis, a navigation correction capability was provided for Apollo 12. A preliminary postflight summary of the descent for Apollo 12, the first pinpoint landing, is also included.

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INTRODUCTION

Lunar Module (LM) descent and ascent premission planning for landing men on the moon started in 1962 with the decision to use the lunar orbit rendezvous (LOR) technique for the Apollo mission (ref. 1). The LOR concept advanced by Houbolt and others is defined in references 1 and 2. This technique allowed design of LM systems and trajectory planning to be optimized for orbital descent to and ascent from the lunar surface.

The LM descent was designed to be accomplished in two powered flight maneuvers: a descent orbit insertion (DOI) maneuver and the powered descent maneuver. The DOI maneuver, a short or impulsive-type transfer maneuver, is performed to reduce the orbit altitude from the command and service module (CSM) parking orbit to a lower altitude for efficiency in initiating the longer, more complex powered descent maneuver. The basic trajectory design for the powered descent was divided into three operational phases: an initial fuel-optimum phase, a landing approach transition phase, and a final translation and touchdown phase. The initial trajectory analysis which led to this design was performed by Bennett and Price (ref. 3). In reference 4, Cheatham and Bennett provided a detailed description of the LM descent design strategy. This description illustrates the complex interactions among systems (guidance, navigation and control, propulsion, and landing radar), crew, trajectory, and operational constraints. A more detailed description of the guidance, navigation, and control system is given by Sears (ref. 5). As LM systems changed from design concept to reality and as operational constraints were modified, it was necessary to modify or reshape the descent trajectory; however, the basic three-phase design philosophy was still utilized.

The LM ascent was designed as a single powered flight maneuver to return the crew from the lunar surface (or from an aborted descent) to a satisfactory orbit from which rendezvous with the CSM could be performed. The basic trajectory design for the powered ascent was divided into two operational phases: a vertical rise phase for surface clearance and a fuel-optimum phase for orbit insertion. Thus, the ascent planning was more straightforward than the descent planning (and, because of the lack of atmosphere, simpler than earth launch planning).

The purpose of the present report is to describe the premission operational planning for LM descent and ascent, that is, to describe the bridge from design planning to flight operational status. A discussion of the primary criteria which precipitated the plan for Apollo 11, the first manned lunar landing on July 20, 1969; a comparison of the

real-time situation with this plan; and a discussion of the postflight analysis and its application to Apollo 12 and subsequent missions are included. A preliminary postflight discussion of Apollo 12, the first pinpoint landing, is also included.

The author wishes to acknowledge the members of the Lunar Landing Section of the Landing Analysis Branch (Mission Planning and Analysis Division) who contributed to the generation of much of the data presented in this report, particularly, W. M. Bolt, J. H. Alphin, J. D. Payne, and J. V. West.

PREMISSION PLANNING

Premission planning entails an integration of mission requirements or objectives with systems and crew capabilities and constraints. This integration is time varying since neither mission requirements nor systems performance remain static. This statement has been particularly true of the LM descent and ascent maneuvers which have been 7 years in design and planning.

A major problem in the design of the descent and ascent maneuvers was the lack of a satisfactory flight simulation; that is, these maneuvers could be simulated properly only by actual performance of the first manned lunar landing mission. For this reason, considerable effort has been spent on reliability, redundancy, and flight safety.

In this section, the final evolution of the planning for the descent and ascent maneuvers for Apollo 11, the first manned lunar landing, will be described. A brief description of the pertinent systems, the guidance logic, the operational design phases, the trajectory characteristics, and the ΔV and propellant requirements for each maneuver is provided.

Descent Planning

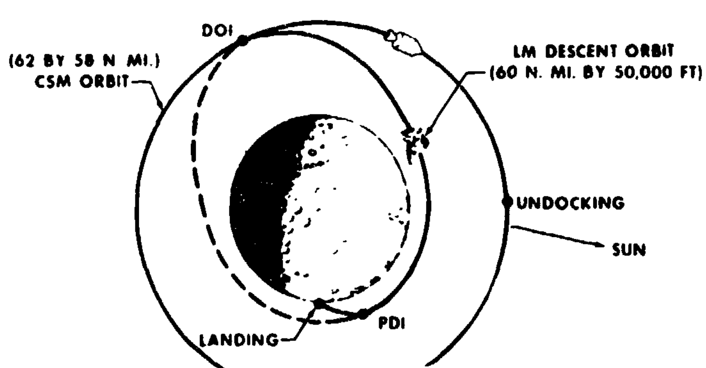
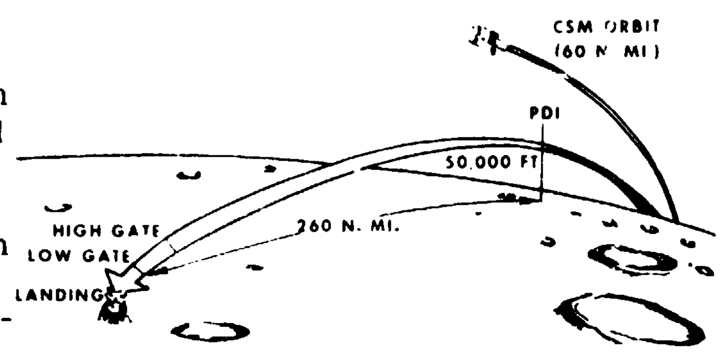


Figure 1. - Lunar module descent.

The LM descent from the CSM parking orbit (approximately 62 by 58 nautical miles) is illustrated in figure 1. After the LM and the CSM have undocked and separated a safe distance (several hundred feet), the LM performs DOI, which is the first and the simplest of the two descent maneuvers. Descent orbit insertion, which is a short retrograde maneuver of approximately 75 fps performed with the descent engine, is made at a position in the orbit 180° (Hohmann-type transfer) from powered descent initiation (PDI), the second descent maneuver. The purpose of the DOI is to efficiently reduce the orbit altitude from approximately 60 nautical miles to 50 000 feet for PDI. Performance of continuous powered descent from altitudes much greater than 50 000 feet is inefficient, and a PDI at lower than 50 000 feet can become a safety hazard (ref. 3). The DOI is described in the operational trajectory

documentation at MSC and is discussed further in the section of this report entitled "Real-Time Analysis." Powered descent planning will be discussed in the remainder of this section.

Operational phases of powered descent. - The LM powered descent trajectory design was established (ref. 1) as a three-phase maneuver (as illustrated in fig. 2) to satisfy the operational requirements imposed on such a maneuver. The first phase, called the braking phase, is designed primarily for efficient propellant usage while reducing orbit velocity and guiding to "high gate" conditions for initiation of the second phase, called the approach phase. The term "high gate" is derived from aircraft pilot terminology for beginning the approach to an airport. The approach phase is designed for pilot visual (out the window) monitoring of the approach to the lunar surface. The final (or landing) phase, which begins at "low gate" conditions (again from pilot terminology), is designed to provide continued visual assessment of the landing site and to provide compatibility for pilot takeover from automatic control for the final touchdown on the surface. A brief description of the systems required and the guidance and targeting logic for achieving these operational phases is given in the following sections. A detailed description of each phase is also given in the operational trajectory documentation.



PHASE	INITIAL EVENT	DESIGN CRITERIA
BRAKING	PDI	MINIMIZE PROPELLANT USAGE
APPROACH	HIGH GATE	CREW VISIBILITY
LANDING	LOW GATE	MANUAL CONTROL

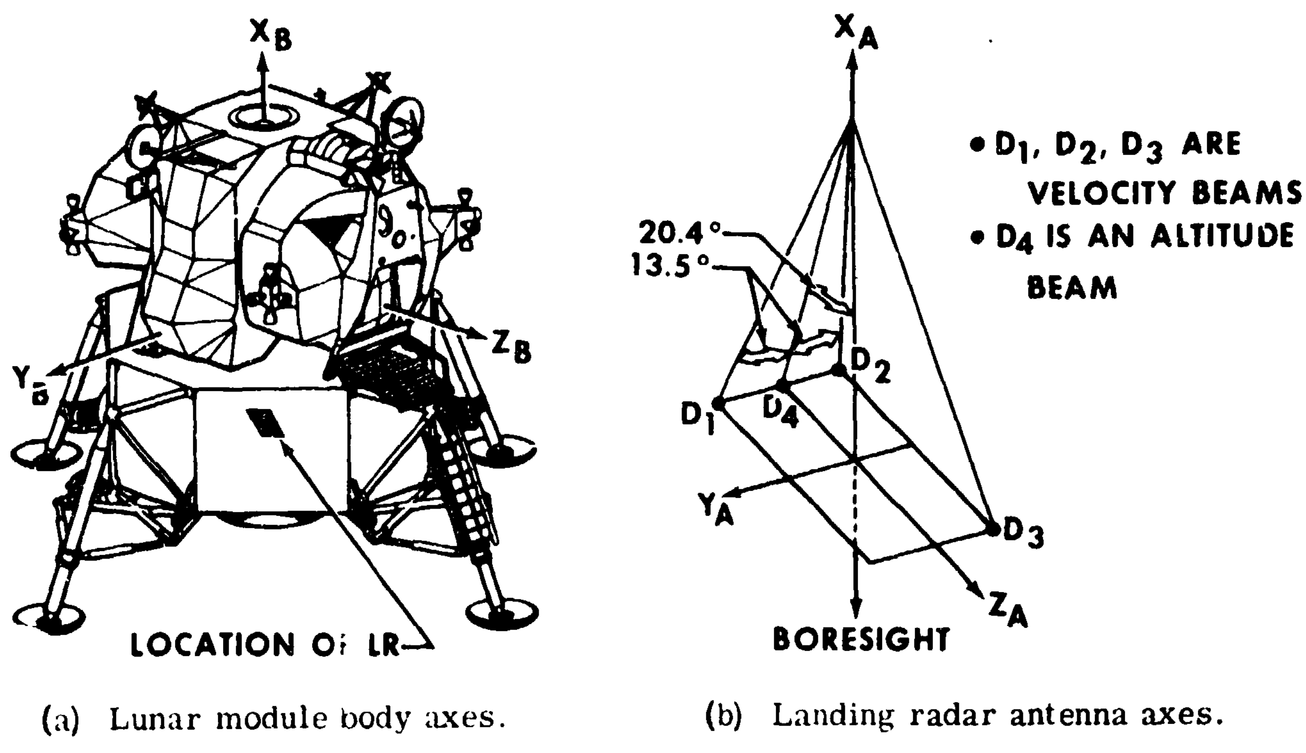
Figure 2. - Operational phases of powered descent.

Systems description. - The success of the LM powered descent is dependent upon the smooth interaction of several systems. The pertinent systems are the primary guidance, navigation, and control system (PGNCS); the descent propulsion system (DPS); the reaction control system (RCS); the landing radar (LR); and the landing point designator (LPD). A detailed description of each system and of the characteristic performance of each system is given in reference 6. A brief description of each system follows.

The PGNCS consists of two major subsystems: an inertial measurement unit (IMU) and a computer. The IMU is the navigation sensor, incorporating accelerometers and gyros to sense changes in velocity and attitude reference. The IMU sends this information to the computer, which contains preprogramed logic for navigation, for calculation of guidance commands, for execution of steering commands (by means of the digital autopilot (DAP)) to the DPS and RCS, for processing of LR measurements of range and velocity relative to the lunar surface, and for display of information to the crew. The crew controls the choice of computer operation through a display and keyboard (DSKY) assembly. A description of the guidance logic is given in a subsequent section. A complete description of the guidance, navigation, and control logic is given in reference 7.

The DPS, containing the rocket engine used for lunar descent and its controls, consists of a throttle and a gimbal drive capable of ±6° of motion. The engine has a maximum thrust of approximately 10 000 pounds (nominal engines varying from 92.5 to 95.5 percent of the design thrust of 10 500 pounds). This thrust level is referred to as the fixed throttle position (FTP) and is used for efficient velocity reduction during the braking phase. It is throttleable between 10 percent and 60 percent for controlled operations in the approach and landing phases. The throttle can be controlled automatically by the PGNCS guidance commands or by manual controls. The gimbal drive is controlled automatically by the DAP for slow attitude rate commands. For high rate changes, the DAP controls the RCS, which consists of four groups of four small control rockets (100 pounds of thrust each) mounted on the LM to control pitch, roll, and yaw.

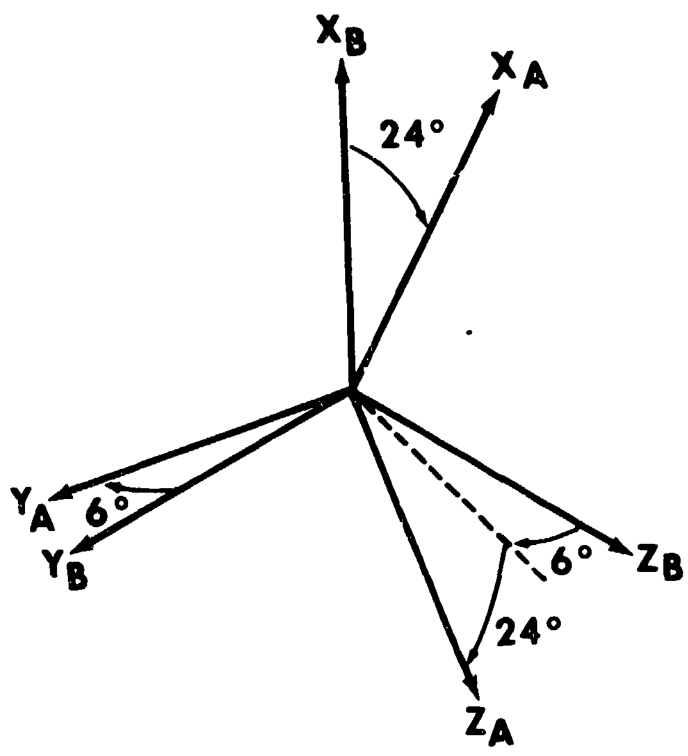
The LR, mounted at the bottom rear of the LM, is the navigation sensor which provides ranging and velocity information relative to the lunar surface. The LR consists of four radar beams, one to provide ranging measurements and three to provide velocity measurements. This beam pattern, which is illustrated relative to the LM body axis system in figure 3, can be oriented in one of two positions, as shown in parts (c) and (d). Position 1 is used in the braking phase when the LM is oriented near the horizontal. Position 2 is used in the approach and landing phases as the LM orientation nears a vertical attitude. The guidance computer converts the ranging information to altitude and updates its navigated state every 2 seconds. The guidance computer also converts the velocity measurement along each beam to platform coordinates and updates a single component of its navigated velocity every 2 seconds (requiring 6 seconds for a complete velocity update). The LR data are also weighted before they are incorporated into the computer (ref. 7).



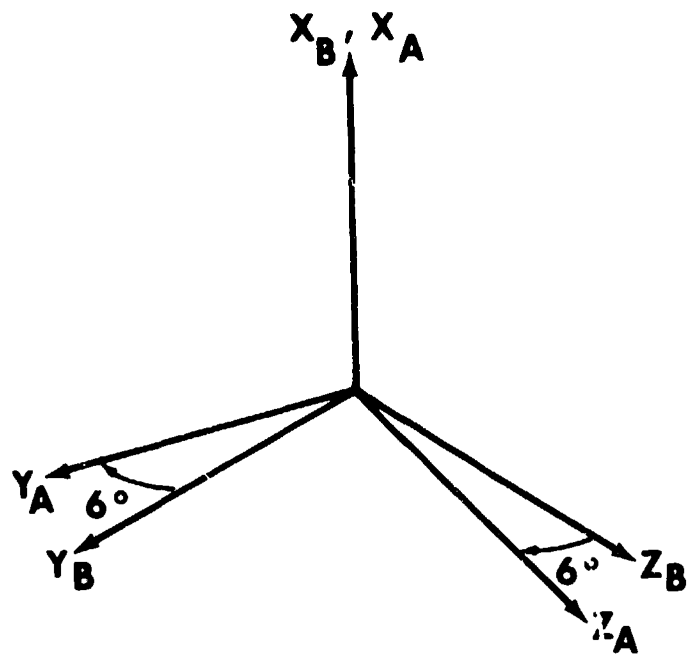
(a) Lunar module body axes.

(b) Landing radar antenna axes.

Figure 3. - Lunar module body and LR axes.



(c) Landing radar position 1 (used in braking phase).



(d) Landing radar position 2 (used on approach and landing phases).

Figure 3. - Concluded.

The final system to be described is a grid on the commander's forward window called the LPD (fig. 4). The window is marked on the inner and outer panes to form an aiming device or eye position. During the approach and landing phases, the computer calculates the look angle (relative to the forward body axis Z_B) to the landing site and displays it on the DSKY. The commander can then sight along the angle on the LPD (zero being along Z_B) to view the landing area to which he is being guided. If the commander desires to change the landing area, he can make incremental changes in plane or cross range by moving the hand controller in the appropriate direction to provide inputs to the computer. Cross-range position is changed in 2° increments, and in-plane position is changed in 0.5° increments. See references 7 and 8 for a detailed description of the guidance logic.

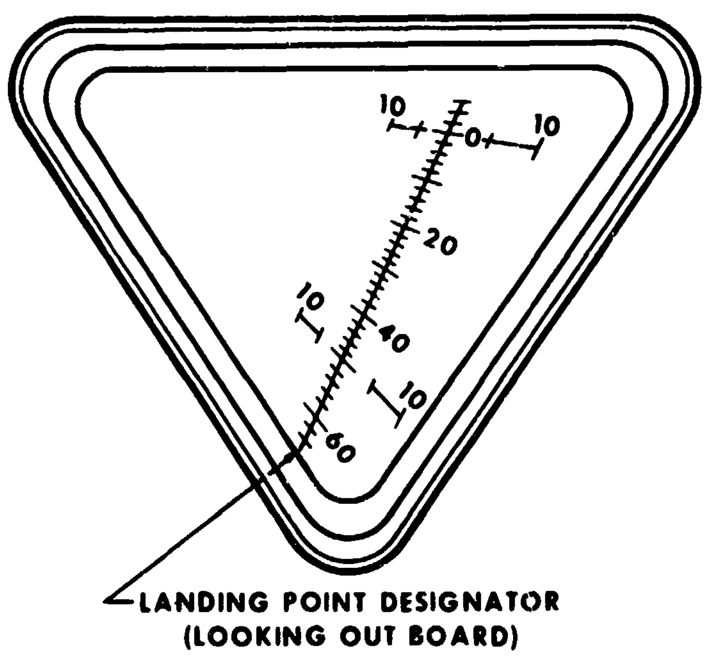


Figure 4. - Forward window.

Guidance logic. - The basic descent guidance logic is defined by an acceleration command which is a quadratic function of time and is, therefore, termed quadratic guidance. A simplified flow chart of this quadratic guidance is given in figure 5. The current LM position and velocity vectors $\vec{R}$ and $\vec{V}$ are determined from the navigation routine. The desired (or target) position vector $\vec{R}_D$, velocity vector $\vec{V}_D$, acceleration vector $\vec{A}_D$, and down-range component of jerk J_{DZ} are obtained from the stored memory. (Jerk is the time derivative of acceleration.) The down-range (horizontal) components of these state vectors (current and desired) are used in the jerk equation to determine time to go (TGO), that is, the time to go from current to desired conditions. If the TGO, the current state, and the desired state are known, then the commanded acceleration vector $\vec{A}_C$ is determined from the quadratic guidance law. It should be noted that the acceleration command equation yields infinite commands when TGO reaches zero. For this reason, the targeting is biased such that desired conditions are achieved prior to TGO reaching zero. Using spacecraft mass M, calculating the acceleration differential between commanded and lunar gravity $\vec{G}$, and applying Newton's law yields a commanded thrust vector $\vec{T}_C$. The magnitude of the vector is used to provide automatic throttling of the DPS. When the throttle commands exceed the throttle region of the DPS (10 to 60 percent), maximum thrust (FTP) is applied. The vector direction is used by the DAP to orient the DPS thrust, by either trim gimbal attitude commands or RCS commands to reorient the entire spacecraft.

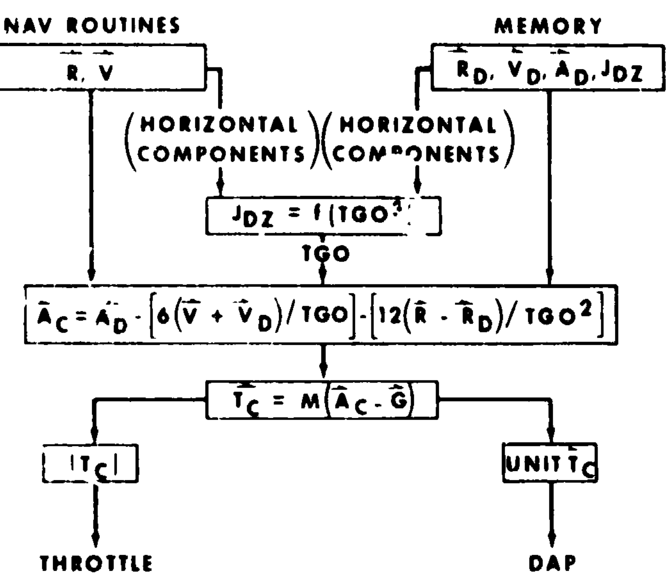


Figure 5. - Basic descent guidance logic.

During the powered descent, the guidance computer provides several sequential programs (P-63 to P-67) for guidance and control operations. A description of each program follows. A complete description of the descent guidance logic and guidance modes is given in references 7 to 9. The first program is P-63 entitled "braking phase guidance." Program P-63 contains an ignition algorithm and the basic guidance logic. The ignition logic determines the time for the crew to ignite the DPS for PDI, based on a stored (preselected) surface range to the landing site. After ignition, the basic guidance logic is used to steer to the desired conditions for beginning the approach phase. As stated previously, the targets are selected with a bias such that the desired conditions are achieved prior to TGO reaching zero. When TGO reaches a preselected value, the guidance program switches automatically from program P-63 to program P-64 entitled "approach phase guidance." This program contains the same basic guidance logic, but a new set of targets. These targets are selected to provide trajectory shaping throughout the approach and landing phases and to establish conditions for initiating an automatic vertical descent from a low altitude to touchdown. In

addition, program P-64 provides window pointing logic for the LPD operation. That is, the landing point will be maintained along the LPD grid on the commander's window. During this time, the crew can make manual inputs with the attitude hand controller to change incrementally (down range or cross range) the intended landing site and remain in automatic guidance. (See the section of this report entitled "Systems Description.")

Again, when TGO reaches a preselected value, the guidance program switches automatically from program P-64 to program P-65 entitled "velocity nulling guidance." This program nulls all components of velocity to preselected values and is used for an automatic vertical descent to the surface, if desired. No position control is used during this guidance mode. The sequencing for automatic guidance is illustrated in figure 6.

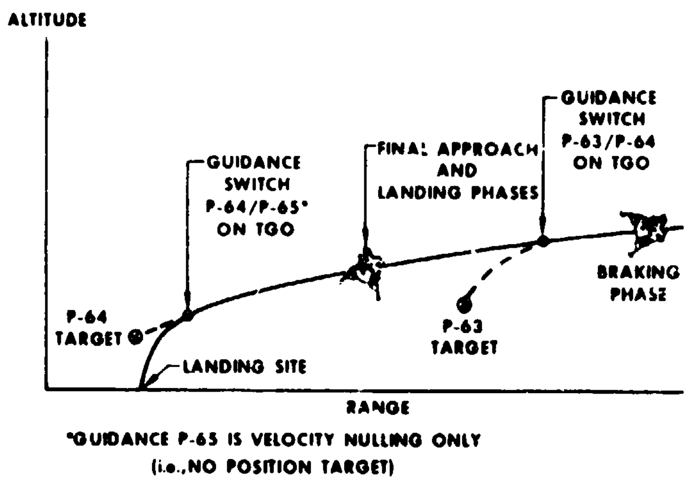


Figure 6. - Target sequence for automatic descent guidance.

Program P-66 entitled "rate of descent" and program P-67 entitled "manual guidance" are optional modes which can be used at crew discretion (manually called up through DSKY) at any time during the automatic guidance modes (programs P-63, P-64, or P-65). During P-66 operation, the crew controls spacecraft attitude, and the computer commands the DPS throttle in order to maintain the desired altitude rate. The desired altitude rate can be adjusted by manual inputs from the crew. This mode is normally entered late in P-64 operation (near low gate) prior to P-65 switching for manual control of the final touchdown position. Program P-67 maintains navigation and display operations for complete manual control of the throttle and attitude. Normally, this mode is not used unless program P-66 is inoperative.

Braking phase. - A scale drawing of the LM powered descent for the Apollo 11 mission is given in figure 7. The intended landing area (designated Apollo site 2) in the Sea of Tranquility is centered at latitude 0.6° N and longitude 23.5° E. The major events occurring during the braking phase (illustrated in fig. 7 and tabulated in table I) are discussed as follows. The braking phase is initiated at a preselected range (approximately 260 nautical miles) from the landing site near perilune of the descent transfer orbit (altitude of approximately 50 000 feet). This point is PDI, which coincides with DPS ignition. Ignition is preceded by a 7.5-second RCS ullage burn to settle the DPS propellants. The DPS is ignited at trim (10 percent) throttle. This throttle setting is held for 26 seconds to allow the DPS engine gimbal to be alined (or trimmed) through the spacecraft center of gravity before throttling up to the maximum, or fixed throttle, position. The braking phase is designed for efficient reduction of orbit velocity (approximately 5560 fps) and, therefore, uses maximum thrust for most of the phase; however, the DPS is throttled during the final 2 minutes of this phase for guidance control of dispersions in thrust and trajectory. As stated earlier, the DPS is throttleable only between 10 and 60 percent; therefore, during FTP operation, the guidance is targeted such that the commanded quadratic acceleration (and consequently the commanded thrust) is a decreasing function. When the command decreases to 57 percent

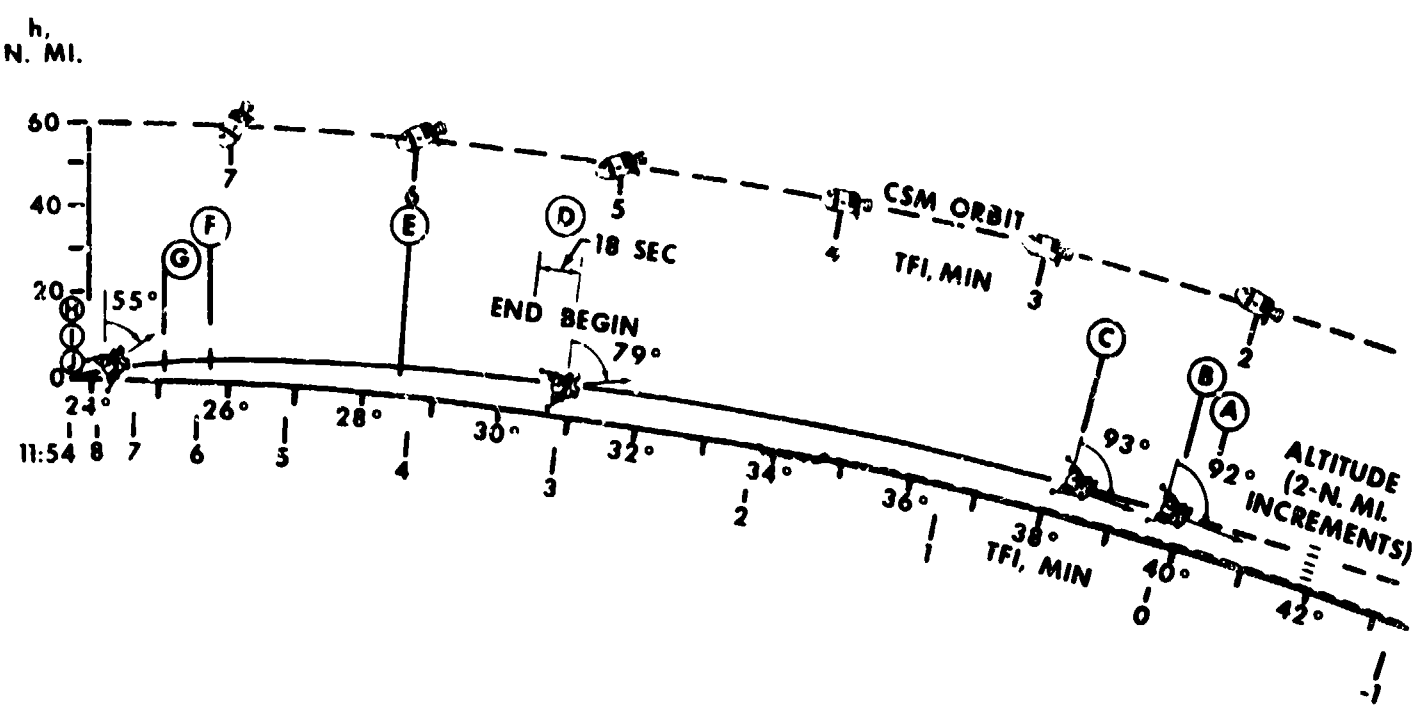


Figure 7. - Apollo 11 LM powered descent.

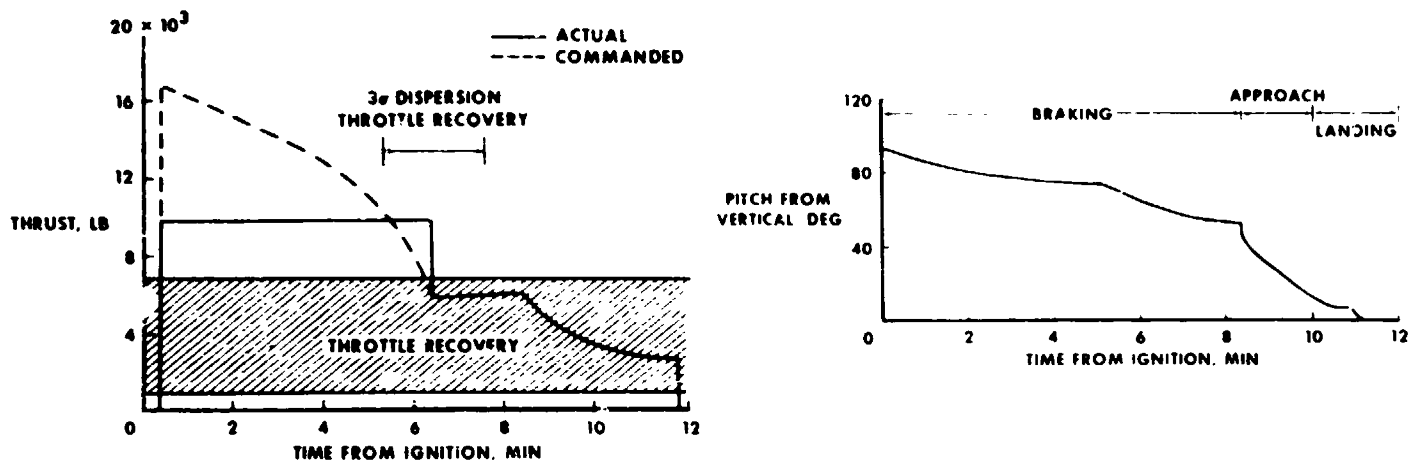
TABLE I. - APOLLO 11 PREMISSION POWERED DESCENT EVENT SUMMARY

Event	TFI,[a] min:sec	Inertial velocity, fps	Altitude rate, fps	Altitude, ft	ΔV, fps
A Ullage	-00:07				
B Powered descent initiation	00:00	5560	-4	48 814	0
C Throttle to maximum thrust	00:26	5529	-3	48 725	31
D Rotate to windows-up position	02:56	4000	-50	44 934	1572
E LR altitude update	04:18	3065	-89	39 201	2536
F Throttle recovery	06:24	1456	-106	24 639	4239
G LR velocity update	06:42	1315	-127	22 644	4399
H High gate	08:26	506	-145	7 515	5375
I Low gate	10:06	55 ([b]68)	-16	512	6176
J Touchdown (probe contact)	11:54	-15 ([b]0)	-3	12	6775

[a]Time from ignition of the DPS.

[b]Horizontal velocity relative to surface.

(a 3-percent low bias), the DPS is throttled as commanded (illustrated by the time history of commanded and actual thrust shown in fig. 8(a)). The thrust attitude (pitch) profile is shown in figure 8(b). Early in the descent, orientation about the thrust axis is by pilot discretion. The Apollo 11 crew oriented in a windows-down attitude for visual ground tracking as a gross navigation check. Rotation to a windows-up attitude is performed at an altitude of approximately 45 000 feet so that the LR can acquire the lunar surface in order to update the guidance computer estimates of altitude and velocity. Altitude updating is expected to begin at an altitude of approximately 39 000 feet. Velocity updating is expected to begin at approximately 22 000 feet.



(a) Thrust. (b) Attitude.

Figure 8. - Time history of thrust and attitude.

The braking phase is terminated when the guidance-calculated TGO (to achieve targets) is reduced to 60 seconds. Termination occurs at an altitude of approximately 7000 feet, a range of approximately 4.5 nautical miles from the landing site, and a time from ignition (TFI) of 8 minutes 26 seconds. The guidance computer automatically switches programs and targets from program P-63 to program P-64 in order to begin the approach phase, as explained in the previous section.

Approach phase. - The approach phase (fig. 9) provides visual monitoring of the approach to the lunar surface. That is, the guidance (program P-64) is targeted to provide spacecraft attitudes and flight time adequate to permit crew visibility of the landing area through the forward window throughout the approach phase. At high gate, in addition to the guidance program switch, the LR antenna is switched from position 1 to position 2 for operation near the surface. (See the section of this report entitled "Systems Description.") The trajectory approach angle (glide angle) is shown to be approximately 16° relative to the surface. This angle allows the crew visual line of sight to the landing area to be above the sun angle (10.9° nominal to 13.6° maximum) even in dispersed (up to 3σ) situations. The angle above the sun line is desirable because surface features tend to be washed out when looking along or below the sun line. (See ref. 10.) The LM attitude, LPD angle, and LR beam geometry are also shown in figure 9. During the approach phase, the altitude decreases from 7000 to 500 feet, the range decreases from approximately 4.5 nautical miles to 2000 feet, and the time of flight is approximately 1 minute 40 seconds. Although no guidance changes or other

transients are made, operationally, the approach phase is considered to be terminated at an altitude of 500 feet (low gate), at which point the landing phase begins.

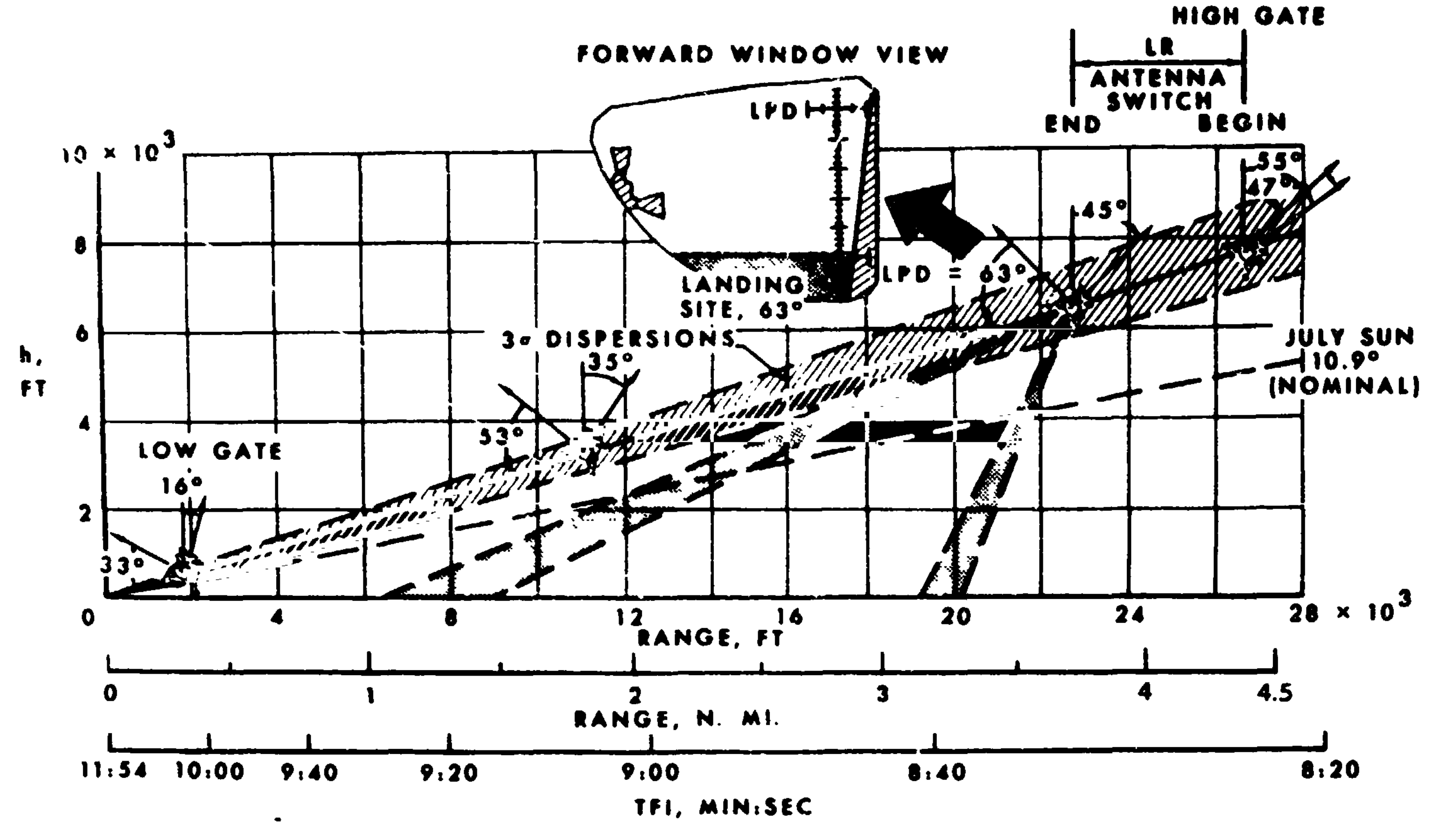


Figure 9. - Approach phase.

Landing phase. - The landing phase is designed to provide continued visual assessment of the landing site and to provide compatibility for pilot takeover from the automatic control. No change occurs in guidance law or targets at this point (low gate) because the approach phase targets have been selected to satisfy the additional constraints. The approach and landing phase targets (program P-64) yield conditions for initiating the automatic vertical descent from an altitude of approximately 150 feet at a 3-fps vertical downward altitude rate. These conditions, along with the selected acceleration and jerk targets, yield trajectory conditions at a 500-foot altitude of 60 fps of forward velocity, 16 fps of vertical rate, and an attitude of approximately 16° off the vertical. These conditions were considered satisfactory by the crew for takeover of manual control. Should the crew continue on automatic guidance, at a TGO of 10 seconds, program P-65 (the velocity nulling guidance) is automatically called to maintain the velocities for vertical descent to the surface. Probes (extended 5.6 feet below the landing pads), upon making surface contact, activate a light which signals the crew to shut down the DPS manually, whether using automatic or manual guidance. The landing phase trajectory is shown under automatic guidance in figure 10.

Premission estimates of dispersions in landing position are shown in figure 11. These dispersions, which are based on a Monte Carlo analysis, include all known systems performance as defined in reference 6. Based on this analysis, the 99-percent-probability landing ellipse was determined to be ± 3.6 nautical miles in plane by ± 1.3 nautical miles cross range.

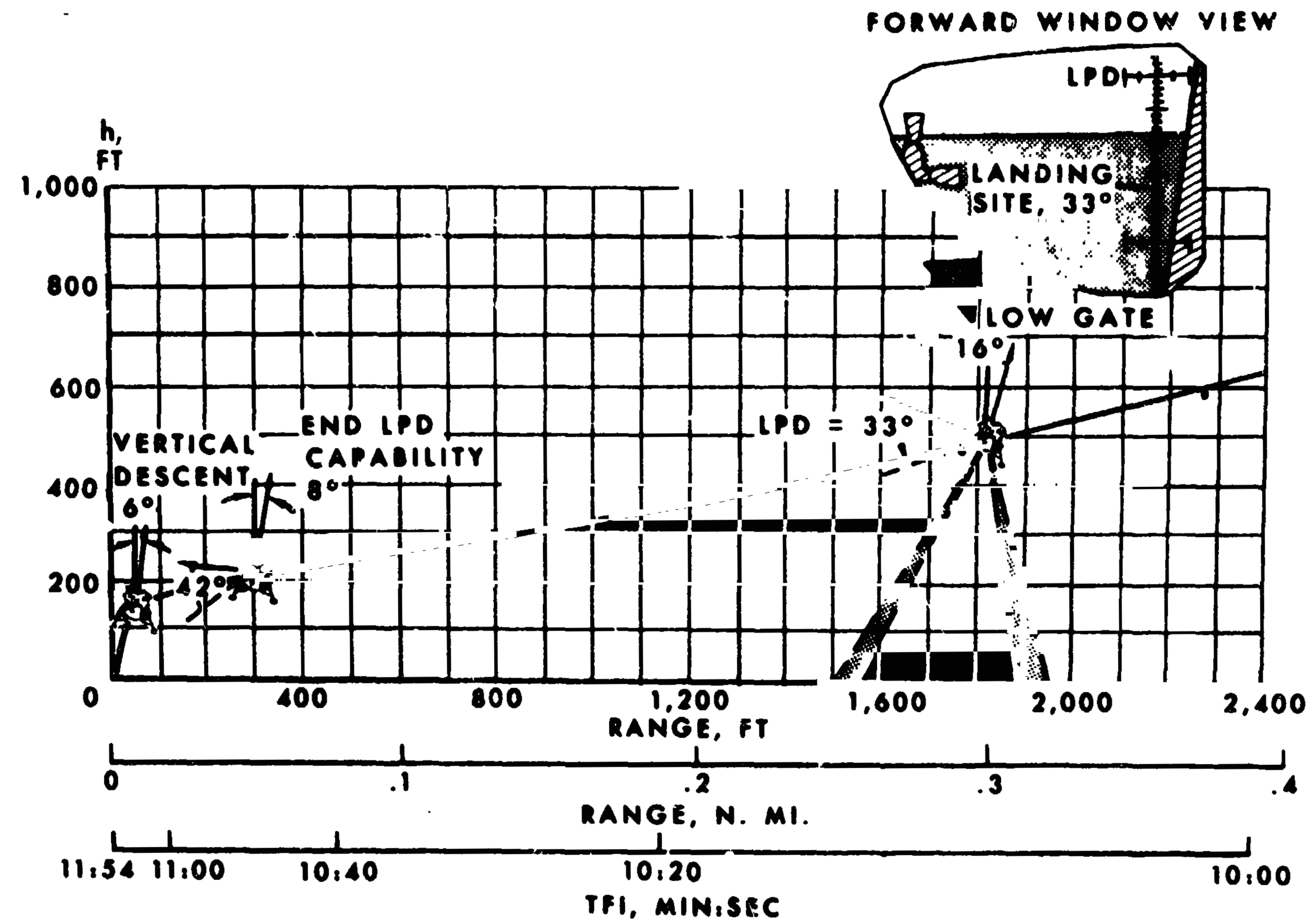


Figure 10. - Landing phase.

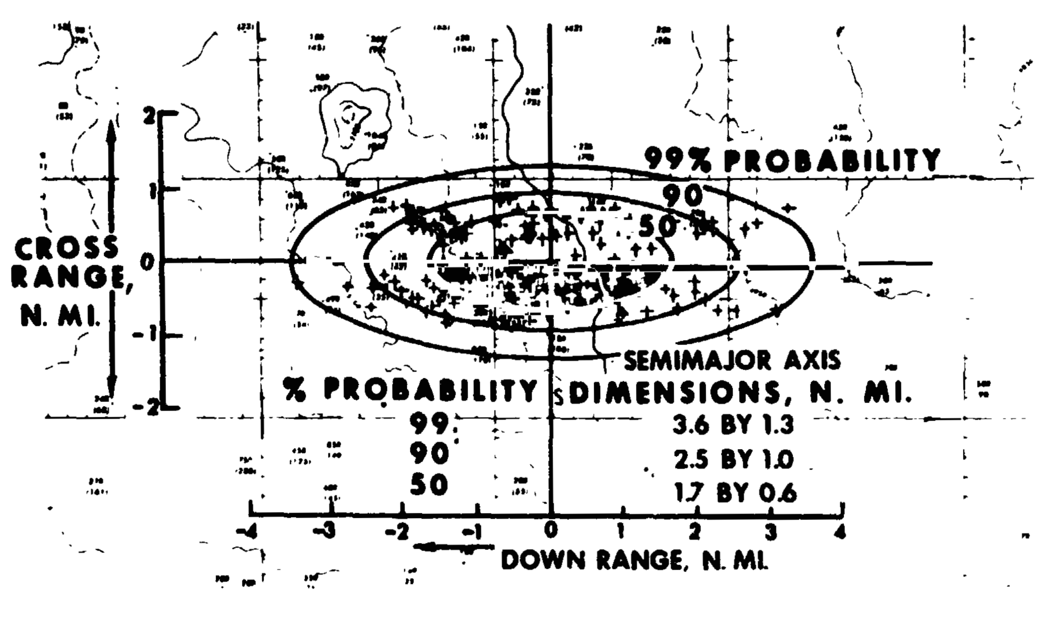


Figure 11. - Predicted Apollo 11 landing dispersions.

The ΔV and propellant requirements. - The ΔV and propellant requirements are determined by the nominal trajectory design, contingency requirements, and dispersions. Consequently, these requirements have undergone continual change. The final design requirements are reported in reference 11. The final operation requirements are given in table II. The required 6827-fps ΔV is established by the automatic guided nominal. In addition, 85 fps is added to assure 2 minutes of flying time in the landing phase, that is, below an altitude of 500 feet. (The automatic guidance requires only 104 seconds of flying time for the landing phase.) Also, a 60-fps ΔV is added for LPD operation in the approach phase in order to avoid large craters (1000 to 2000 feet in diameter) in the landing area. Contingency propellant allotments are provided for failure of a DPS redundant propellant flow valve and for bias on propellant low-level light operation. The valve failure causes a shift in propellant mixture ratio and a lower thrust (by about 160 pounds), but otherwise, DPS operation is satisfactory. The low-level light signifies approaching propellant depletion; therefore, a bias is used to protect against dispersions in the indicator. If the low-level light should fail, the crew uses the propellant gage reading of 2 percent remaining as the abort decision indicator. The light sensor provides more accuracy and is therefore preferred over the

TABLE II. - DESCENT ΔV AND PROPELLANT REQUIREMENTS

Item	Propellant required, lb	Propellant remaining, lb
System capacity (7051.2 lb fuel, 11 209.3 lb oxidizer)	--	18 260.5
Offloaded (minimize malfunction penalty)	75.4	18 185.1
Unusable	250.5	17 934.6
Available for ΔV		17 934.6
Nominal required for ΔV (6827 fps)	16 960.9	973.7
Dispersions (-3σ)	292.0	681.7
Pad	--	681.7
Contingencies		
Engine valve malfunction ($\Delta MR = \pm 0.016$)	64.7	617.0
Redline low-level sensor	68.7	548.3
Redesignation (60 fps)	102.9	445.4
Manual hover (85 fps)	144.0	301.4
Margin	--	301.4

gage reading. The ground flight controllers call out time from low-level light ON in order to advise the crew of impending propellant depletion for an abort-or-landing decision point at least 20 seconds prior to depletion. This procedure allows the crew to start arresting the altitude rate with the DPS prior to an abort stage to prevent surface impact. The allowance for dispersions is determined from the Monte Carlo analysis mentioned previously. As can be seen from table II, the ΔV and propellant requirements are satisfied by a positive margin of 301 pounds. This margin can be converted to an additional hover or translation time of 32 seconds.

Ascent Planning

A sketch of the LM ascent from the lunar surface is given in figure 12. The ascent has a single objective, namely, to achieve a satisfactory orbit from which rendezvous with the orbiting CSM can subsequently be performed. Nominally, insertion into a 9- by 45-nautical-mile orbit, at a true anomaly of 18° and an altitude of 60 000 feet, is desired. The time of lift-off is chosen to provide the proper phasing for rendezvous. Not the choice of targeting for rendezvous, but rather a description of the powered ascent only, is the subject of this section.

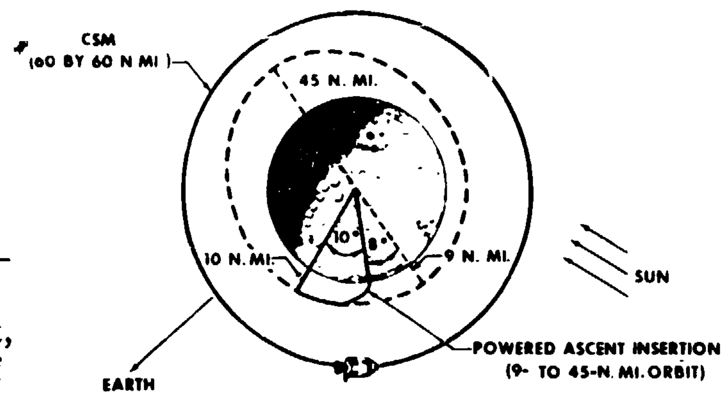


Figure 12. - Lunar module ascent.

Systems description. - Only three pertinent systems are required for ascent — the PGNCS and the RCS, which have already been described, and the ascent propulsion system (APS). The APS, unlike the DPS, is not throttleable and does not have a trim gimbal drive, but provides a constant thrust of approximately 3500 pounds throughout the ascent (ref. 6). Engine throttling is not required during ascent, since down-range position control is not a target requirement; that is, only altitude, velocity, and orbit plane are required for targeting. This thrust can be enhanced slightly (by approximately 100 pounds) by the RCS attitude control. The ascent DAP logic is such that only +X body axis (along the thrust direction) jets are fired for attitude control during ascent.

A fourth system, the abort guidance system (AGS), should also be mentioned. The AGS is a redundant guidance system to be used for guidance, navigation, and control for ascent or aborts in the event of a failure of the PGNCS. The AGS has its own computer and uses body-mounted sensors instead of the inertial sensors as used in the PGNCS. A detailed description of the AGS is given in references 12 and 13.

Operational phases. - The powered ascent is divided into two operational phases: vertical rise and orbit insertion. The vertical rise phase is required for the ascent stage in order to achieve terrain clearance. (The trajectory for propellant optimization takes off along the lunar surface.) A description of trajectory parameters and LM attitude during the vertical rise phase and during the transition to the orbit insertion phase is shown in figure 13. The guidance switches to the orbit insertion phase when the radial rate becomes 40 fps. However, because of DAP steering lags, the pitchover does not begin until a radial rate of approximately 50 fps is achieved. This delay

means that the vertical rise phase is terminated 10 seconds after lift-off. Also, during the vertical rise, the LM Z body axis is rotated to the desired azimuth, which is normally in the CSM orbit plane.

The orbit insertion phase is designed for efficient propellant usage to achieve orbit conditions for subsequent rendezvous. The orbit insertion phase, the total ascent phase performance, insertion orbit parameters, and onboard displays at insertion are shown in figure 14. The onboard display values reflect the computer-estimated values. Yaw steering is used during the orbit insertion phase, if required, to maneuver the LM into the CSM orbit plane or into a plane parallel with the CSM orbit. In the nominal case, no yaw steering is required. The nominal ascent burn time is 7 minutes 18 seconds with a 3σ dispersion of $\pm$ 17 seconds. The trajectory dispersions are plotted in figure 15. The ascent guidance logic is discussed in the following section.

Guidance logic. - The ascent guidance logic commands only attitude since no engine throttling is required. For the vertical rise phase, the logic is simple: the initial attitude is held for 2 seconds in order to clear the descent stage; the attitude is pitched to the vertical while rotating to the desired azimuth; and termination occurs when the altitude rate is greater than or equal to 40 fps upward or when the altitude is greater than 25 000 feet (used for aborts off descent).

The insertion phase guidance logic is defined by an acceleration command which is a linear function of time and is, therefore, termed linear guidance. The TGO is determined as a function of velocity to be gained, that is, the difference between current and desired velocity. This TGO, along with current and desired targets, is used to determine acceleration commands in radial and cross-range directions. The acceleration available from the APS is oriented by firing the RCS

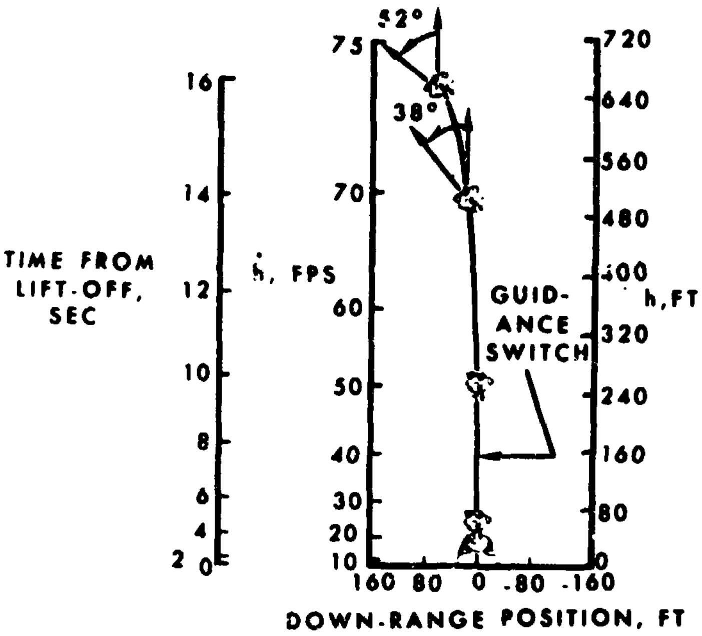


Figure 13. - Vertical rise phase.

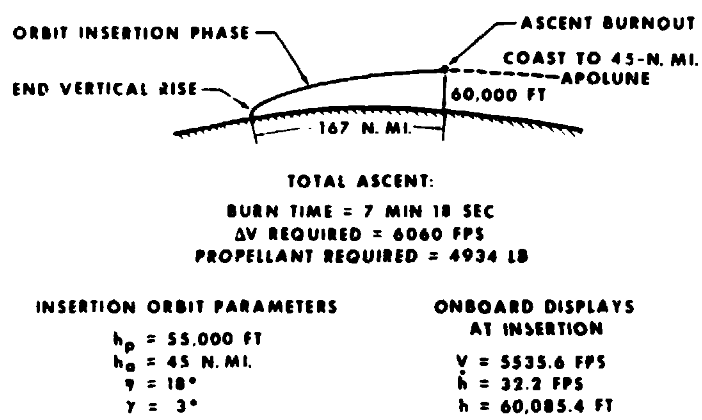


Figure 14. - Orbit insertion phase.

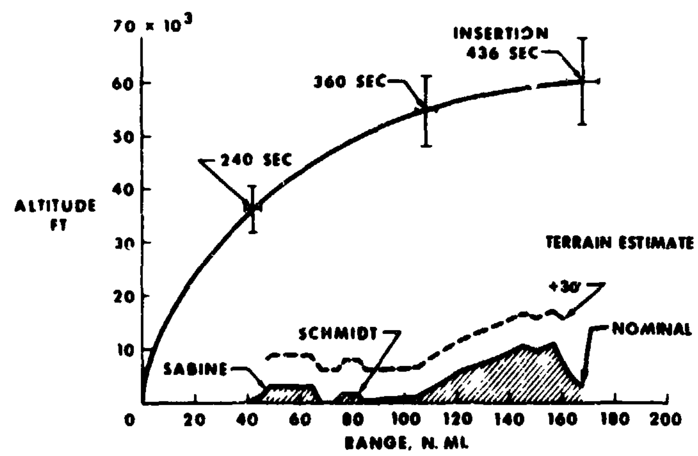


Figure 15. - Predicted Apollo 11 ascent dispersions.

according to the DAP logic to satisfy these commands, with any remaining acceleration being applied in the down-range direction. Cross-range steering is limited to 0.5°. Out-of-plane maneuvering greater than 0.5° is combined with the subsequent rendezvous sequencing maneuvers. When TGO becomes less than 4 seconds, a timer is activated to cut off the APS at that time.

Three ascent guidance programs are used: program P-12 for ascent from the surface, program P-70 for ascent aborts during descent (to be performed with the DPS), and program P-71 for ascent aborts during descent (to be performed with the APS). All the programs use the vertical rise and insertion logic described previously. They differ only by the targeting logic used to establish the desired orbit insertion conditions. For aborts at PDI and through the braking phase, the LM (as a result of the DOI maneuver) is ahead of the CSM. During the approach and landing phases, the CSM moves ahead of the LM. Therefore, the desired orbit insertion conditions targeted by programs P-70 and P-71 vary as a function of phase relationship between the LM and CSM to establish rendezvous sequencing. Reference 7 contains a complete description of the ascent guidance logic.

The ΔV and propellant requirements. - The ΔV and propellant requirements are determined by the nominal trajectory design, contingency requirements, and dispersions. Consequently, the requirements for ascent, as for descent, have undergone continual change. The final design requirements are given in reference 11. The final operation requirements are given in table III. The required 6056-fps ΔV is established by the nominal insertion into a 9- by 45-nautical-mile orbit. In addition, a 54-fps ΔV is provided for two contingencies. A 40-fps ΔV is provided for the first contingency, which is a switchover from PGNCS to AGS for inserting from an off-nominal trajectory caused by a malfunctioning PGNCS. A 14-fps ΔV is provided for the second contingency, in which the thrust-to-weight ratio is reduced in an abort from a touchdown situation wherein the LM is heavier than the nominal lift-off weight. (Some weight is nominally off-loaded on the lunar surface.) Also, 19 pounds of propellant is allotted for contingency engine valve malfunction as in the descent requirements. The allowance for dispersions is determined from the Monte Carlo analysis. As can be seen from tabie III, the ΔV and propellant requirements are satisfied, with a positive margin of 48 pounds.

TABLE III - ASCENT ΔV AND PPOPELLANT REQUIREMENTS

Item	Propellant required, lb	Propellant remaining, lb
System capacity (2026.0 lb fuel, 3218.4 lb oxidizer)	--	5244.4
Offloaded (minimize malfunction penalty)	20.7	5223.7
Unusable	56.3	5167.4
Available for ΔV	--	5167.4
Nominal required for ΔV (6055.[illegible] fps)	4966.7	200.7
Dispersions (-3σ)	66.7	134.0
Pad	--	134.0
Contingencies		
Engine valve malfunction ($\Delta MR = \pm 0.016$)	18.8	115.2
PGNCB to AGS switchover (40 fps)	23.8	91.4
Abort from touchdown ($\Delta W = +112.9$ lb, $\Delta(\Delta V) = -14.3$ fps)	43.2	48.2
Margin	--	48.2

REAL-TIME ANALYSIS

During the real-time situation, monitoring of the spacecraft systems and of the trajectory is performed continually both on board by the crew and on the ground by the flight controllers. This monitoring determines whether the mission is to be continued or aborted as established by mission techniques prior to flight. The real-time situation for Apollo 11 descent and ascent is described in the following section.

Descent Orbit Insertion

The DOI maneuver is performed on the far side of the moon (at a position in the orbit 180° prior to PDI) and is, therefore, executed and monitored solely by the crew. Of major concern during the burn is the performance of the PGNCS and the DPS. The DOI maneuver is essentially a retrograde burn to reduce orbit altitude from approximately 60 nautical miles to 50 000 feet for PDI and requires a ΔV reduction of 75 fps. This reduction is accomplished by throttling the DPS to 10-percent thrust for 15 seconds (c.g. trimming) and to 40-percent thrust for 13 seconds. An overburn of 12 fps (or 3 seconds) would cause the LM to be on an impacting trajectory prior to PDI. Thus, the DOI is monitored by the crew with the AGS during the burn and by range-rate tracking with the rendezvous radar (RR) immediately after the burn. If the maneuver is unsatisfactory, an immediate rendezvous with the CSM is performed with the AGS. For Apollo 11, this maneuver was nominal. (Down-range residuals after the burn were 0.4 fps.)

Powered Descent

The powered descent is a complex maneuver which is demanding on both crew and systems performance. Therefore, as much monitoring as possible is performed on the ground in order to reduce crew activities and to use sophisticated computing techniques not possible on board. Obviously, time-critical failures and near-surface operations must be monitored on board by the crew for immediate action. Pertinent aspects of guidance, propulsion, and flight dynamics real-time monitoring of the powered descent are given as follows.

The PGNCS monitoring. - To determine degraded performance of the PGNCS, the ground flight controllers continually compare the LM velocity components computed by the PGNCS with those computed by the AGS and with those determined on the ground through Manned Space Flight Network (MSFN) tracking. That is, a two-out-of-three voting comparison logic is used to determine whether the PGNCS or the AGS is degrading. The powered flight processor used to compute LM velocity from MSFN tracking data is explained in reference 14. Limit or redlines for velocity residuals between the PGNCS and the MSFN computations and between the PGNCS and the AGS computations are established premission, based on the ability to abort on the PGNCS to a safe (30 000-foot perilune) orbit.

In real time, the Apollo 11 PGNCS and AGS performance was close to nominal; however, a large velocity difference between the PGNCS and the MSFN computations in the radial direction of 18 fps (limit line is 35 fps) was detected at PDI, remaining constant well into the burn. This error did not indicate a systems performance problem, but rather an initialization error in down-range position. This effect is illustrated geometrically in figure 16. The PGNCS position $\vec{R}_E$ and velocity $\vec{V}_E$ estimates are used to initiate the MSFN powered flight processor. The MSFN directly senses the actual velocity $\vec{V}_A$ at the actual position $\vec{R}_A$, but being initialized by the PGNCS state, applies $\vec{V}_A$ at $\vec{R}_E$. Thus, a flight-path-angle error $\Delta\gamma$ is introduced by a down-range-position error and shows up as a radial velocity difference $\Delta\vec{V}_{DIFF}$. The magnitude of the velocity difference indicates that the Apollo 11 LM down-range position was in error by approximately 3 nautical miles at PDI and throughout the powered descent to landing. The reason for the down-range navigation error was attributed to several small ΔV inputs to the spacecraft state in coasting flight. These inputs were from uncoupled RCS attitude maneuvers and cooling system venting not accounted for by the propagation of the predicted navigated state at PDI.

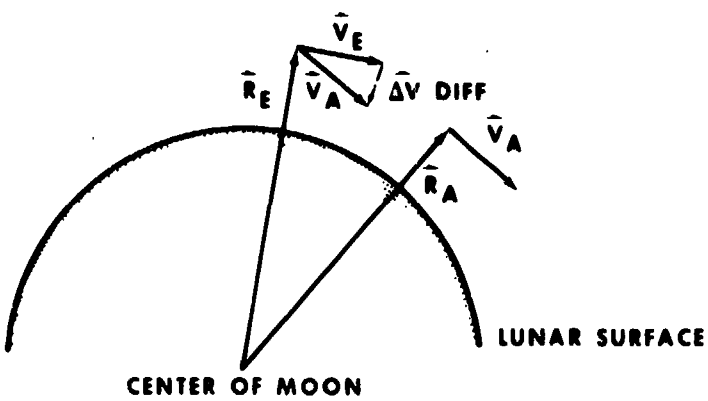


Figure 16. - Effect of position error on velocity comparison.

The LM guidance computer (LGC) also monitors the speed at which it is performing computation tasks (navigation, guidance, displays, processing radar data, and auxiliary tasks). If the computer becomes overloaded or falls behind in accomplishing these tasks, an alarm is issued (to inform crew and flight controllers), and priorities are established so that the more important tasks are accomplished first. This alarm system is termed computer restart protection. During real time, an erroneous voltage signal from the RR was sent to the computer. This signal caused the computer to continually calculate angles from RR tracking of the CSM and consequently to fall behind in completing its tasks. As a result, the alarm was displayed, and computation priorities were executed by the computer. The alarm was quickly recognized, and flight control monitoring indicated that guidance and navigation functions were being performed properly; thus, the descent was continued. Despite the initial position error and the RR inputs, the PGNCS performed excellently during powered descent of the Apollo 11 mission.

The DPS-PGNCS interface. - To determine in real time if the DPS is providing sufficient thrust to achieve the guidance targets, the flight controllers monitor a plot of guidance thrust command (GTC) versus horizontal velocity, as shown in figure 17.

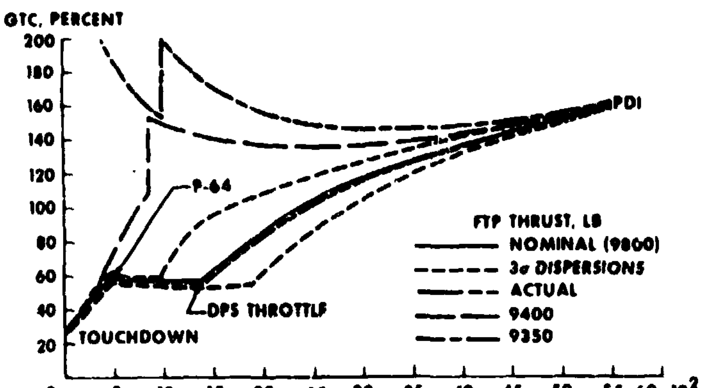


Figure 17. - Guidance thrust command versus horizontal velocity.

Nominally, the GTC decreases (approximately parabolically) from an initial value near 160 percent to the throttleable level 57 percent approximately 2 minutes (horizontal velocity being 1400 fps) prior to high gate (horizontal v_locity being 500 fps). If the DPS produces off-nominal high thrust, horizontal velocity is being reduced more rapidly than desired in order to reach high-gate conditions. Therefore, the GTC drops to 57 percent earlier (at higher than nominal velocity) in order to guide to the desired position and velocity targets. This early throttledown results in propellant inefficiency. If the DPS produces off-nominal low thrust, horizontal velocity is not being reduced rapidly enough. Therefore, the GTC drops to 57 percent later (at lower velocity) in order to guide to the desired position and velocity. This later throttledown results in increased propellant efficiency (i.e., longer operation at maximum thrust). However, if no throttledown occurs prior to high gate (program switch from P-63 to P-64), the targets will not be satisfied, and the resulting trajectory may not be satisfactory (from the standpoint of visibility). In fact, for extremely low thrust, the guidance solution for GTC can diverge (fig. 17); as TGO becomes small, the guidance calls for more and more thrust in order to achieve its targets. This divergence can result in an unsafe trajectory, one from which an abort cannot be satisfactorily performed. The 2-minute bias for throttle recovery prior to high gate provides sufficient margin for 3σ low thrust even with propellant valve malfunction. However, flight controllers monitor GTC to assure satisfactory interface between DPS and PGNCS operation. A mission rule was established that called for an abort based on GTC divergence. During Apollo 11, the DPS thrust was nearly nominal (fig. 17); thus, no DPS-PGNCS interface problems were encountered.

The LR-PGNCS interface. - Normally, LR update of the PGNCS altitude estimate is expected to occur (by crew input) at an altitude of 39 000 ± 5000 feet (3σ dispersion). Without LR altitude updating, systems and navigation errors are such that the descent cannot be safely completed. In fact, it is unsafe to try to achieve high gate (where the crew can visually assess the approach) without altitude updating. Thus, a mission rule for real-time operation was established that called for aborting the descent at a PGNCS-estimated altitude of 10 000 feet if altitude updating has not been established.

In addition to the concern for the time initial altitude updating occurs is the concern for the amount of altitude updating (that is, the difference between PGNCS and LR altitude determinations Δh). If the LM is actually higher than the PGNCS estimate, the LR will determine the discrepancy and update the PGNCS. The guidance then tries to steer down rapidly to achieve the targets. As a result of the rapid changes, altitude rates may increase to an unsafe level for aborting the descent. That is, should an abort be required, the altitude rates could not be nulled by the ascent engine in time to prevent surface collision. The Δh limits necessary to avoid these rates are shown in

figure 18. Notice that over the estimated 3σ region of LR initial updating (which at the time of that analysis was centered at an altitude of only 35 600 feet instead of 39 000 feet), the Δh limits are much greater than the $+3\sigma$ navigation estimates of Δh. However, flight controllers, as well as the crew, monitor Δh to assure the boundary is not exceeded before incorporation of the LR altitude updating. If the boundary is exceeded, then the data are not incorporated, and an abort is called. When the LM is actually lower than estimated, no excessive rates are encountered upon LR updating. It is necessary only that the LM altitude and altitude rate be above the abort limits, which are defined in the section of this report entitled "Trajectory Limits."

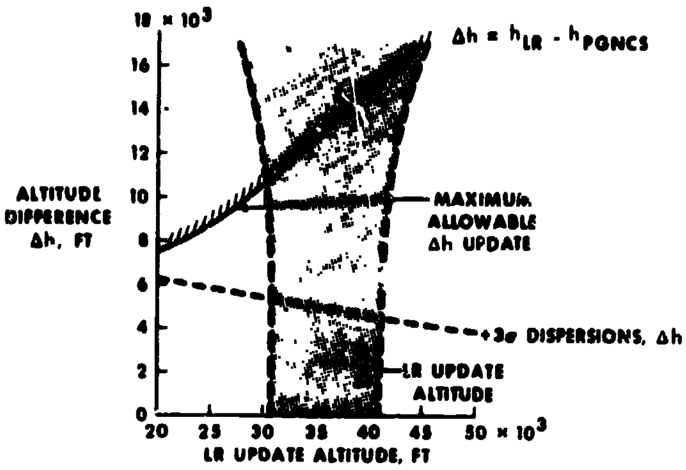


Figure 18. - Landing-radar altitude updates.

During Apollo 11, the LR acquired lock-on to the lunar surface during the rotation to face-up attitude at an altitude of 37 000 feet. The Δh was -2200 feet (indicating that the LM was actually low). This small amount of Δh can readily be attributed to terrain variations. Since no limits were violated, the data were incorporated after a short period of monitoring at an altitude of 31 600 feet. The Δh readily converged to a small value of 100 feet within 30 seconds. The LR velocity updates were incorporated nominally, beginning at a 29 000-foot altitude. As expected, LR signal dropouts were encountered at low altitudes (below 500 feet) but presented no problem. (When the velocity becomes small along the LR beams, depending on the attitude and approach velocity, zero Doppler shift can be encountered; hence, no signal occurs.)

Trajectory limits. - During real time, trajectory limits are monitored for flight safety. The prime criteria for flight safety are the ability to abort the descent at any time until the final decision to commit to touchdown. Thus, flight dynamics limits are placed on altitude and altitude rate, as shown in figure 19. Notice that the nominal trajectory design does not approach the limits until late in the descent, after the crew has had ample time for visual assessment of the situation. The limits shown are based on APS abort with a 4-second free fall for crew action delay or a DPS abort with a 20-second communications delay for ground notification. The flight controllers and the crew monitor altitude and altitude rate, but because of communication delays with the ground, the flight controllers only advise, based on projected trends. The Apollo 11 altitude and altitude rate profile shown in figure 19 was near nominal.

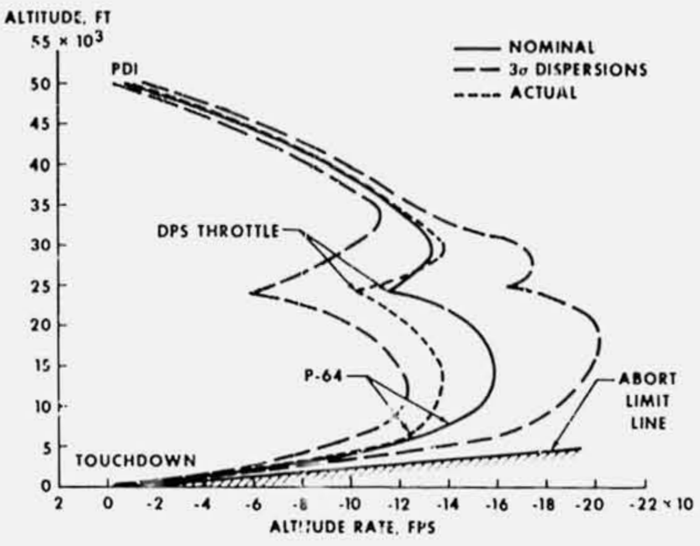


Figure 19. - Altitude versus altitude rate during powered descent.

Crew visual assessment. - As stated previously, the approach and landing phases have been designed to provide crew visibility of the landing area. This provision allows the crew to assess the acceptability of the landing area — to decide to continue toward the landing area or to redesignate (with LPD or manual control) a landing away from it. During Apollo 11, because of the initial navigation errors, the descent was guided into the generally rough area surrounding West Crater (see fig. 20 and the section of this report entitled "The PGNCS Monitoring"). West Crater is inside the premission mapped area approximately 3 nautical miles west of center. Unfortunately, because of the

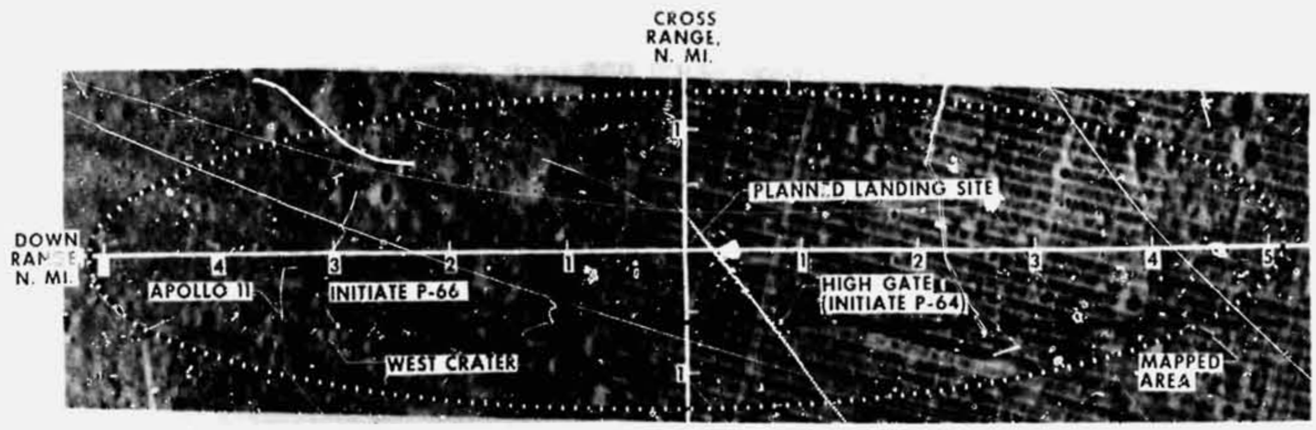


Figure 20. - Apollo 11 landing site.

guidance program alarms, the commander was unable to concentrate on the window view until late in the descent (near low gate). Thus, crew visual assessment during the approach phase was minimal, resulting in continued approach into the West Crater area. This statement is discussed further in the subsequent section entitled "Postflight Analysis."

Ascent

During the real-time situation, the crew and flight controllers continually monitor the systems and trajectory for detection of off-nominal performance. Of primary concern is the performance of the APS and the PGNCS. The APS, of course, must perform, as no backup propulsion system is provided. Should the APS fail during the final 30 seconds of ascent, the RCS can complete the insertion. The PGNCS performance is monitored by the AGS and powered flight processor, using MSFN tracking in the same manner as in the descent guidance monitoring. The limit lines are set for completion of the ascent on the AGS should the PGNCS performance degrade.

In real time, the PGNCS (as well as the AGS) performance was excellent, and guidance switchover was not required. The APS performance was also excellent. Insertion occurred at 7 minutes 15 seconds from lift-off, with 7 minutes 18 seconds being the operational trajectory prediction.

POSTFLIGHT ANALYSIS

Postflight analysis is conducted to determine how the actual flight performance compared with the premission planning. The purpose of postflight analysis is to determine if the premission planning was adequate and, if not, to determine the changes required for subsequent flights. A brief description of the Apollo 11 postflight results for LM descent and ascent, application of these results to Apollo 12 planning and a preliminary postflight estimate of Apollo 12 are given.

Apollo 11 Descent and Ascent

Descent. - The DOI maneuver was performed nominally, as discussed in the preceding section. The events during powered descent are tabulated in table IV. The braking phase events were near nominal (table I). Rotation to a windows-up attitude was delayed slightly because of the crew's selection of a slow rotation rate. This rotation delay resulted in the slight delay in acquiring LR (which was acquired prior to completion of the rotation). The approach phase, as shown in figure 21, also agreed well with premission planning. As shown previously (fig. 20), the descent headed into the area near West Crater because of initial navigation error (approximately 3 nautical miles down range). During the approach phase, the LPD indicated to the commander that the automatic system was guiding to a landing up range of West Crater. Later on, the landing appeared to be heading into the rock field just beyond West Crater. This uncertainty was due to several factors: the time rate of change in LPD angle, errors introduced by terrain variations (primarily slope), and the lack of time for visual assessment because of crew diversion to guidance program alarms. (Refer to the section entitled "Real-Time Analysis.") Therefore, not until the beginning of the landing

TABLE IV. - LUNAR DESCENT EVENT TIMES

g. e. t., [a] hr:min:sec	Event
102:17:17	Acquisition of data
102:20:53	LR on
102:24:40	Alinement of abort guidance to primary guidance
102:27:32	Yaw maneuver to obtain improved communications
102:32:55	Altitude of 50 000 feet
102:32:58	Propellant-settling firing start
102:33:05	Descent engine ignition
102:33:31	Fixed-throttle position (crew report)
102:36:57	Face-up yaw maneuver in process
102:37:51	LR data good
102:37:59	Face-up maneuver complete
102:38:22	1202 alarm (computer determined)
102:38:45	Enabling of radar updates
102:38:50	Altitude less than 30 000 feet (inhibit X-axis override)
102:38:50	Velocity less than 2000 fps (start LR velocity update)
102:39:02	1202 alarm
102:39:31	Throttle recovery
102:41:32	Program P-64 entered
102:41:37	LR antenna to position 2
102:41:53	Attitude hold (handling qualities check)
102:42:03	Automatic guidance
102:42:18	1201 alarm (computer determined)
102:42:19	LR low scale (less than 2500 feet)
102:42:43	1202 alarm (computer determined)
102:42:58	1202 alarm (computer determined)
102:43:09	Landing-point redesignation
102:43:13	Attitude hold
102:43:20	Update of abort guidance altitude
102:43:22	Program P-66 entered
102:44:11	LR data not good
102:44:21	LR data good
102:44:28	Propellant low-level sensor light on
102:44:59	LR data not good
102:45:03	LR data good
102:45:40	Landing
102:45:40	Engine off

[a]Ground elapsed time.

phase did the commander try to avoid the large area of rough terrain by taking over manual control (P-66 guidance) at an altitude of 410 feet when the forward velocity was only 50 fps. An LPD input was made, as shown in table IV; but in discussions with the crew, it was determined that this input was inadvertent. The landing phase is illustrated in figure 22, and the groundtrack is shown in figure 23. The landing site is shown to have been moved, through manual maneuvering, approximately 1100 feet down range and 400 feet cross range from where the automatic guided descent (under P-64/P-65 control) would have landed. The attitude profile and the altitude/altitude-rate profile are shown in figures 24 and 25, respectively. The somewhat erratic behavior of these profiles can best be explained by Commander Neil A. Armstrong's comments to the Society of Experimental Test Pilots meeting in Los Angeles on September 26, 1969, "I [was] just absolutely adamant about my God-given right to be wishy-washy about where I was going to land."

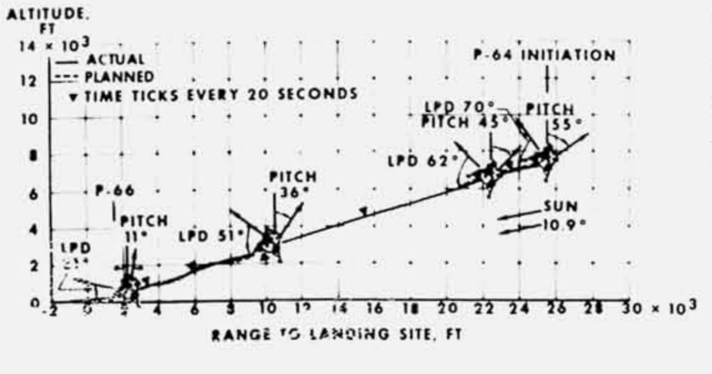


Figure 21. - Apollo 11 approach phase.

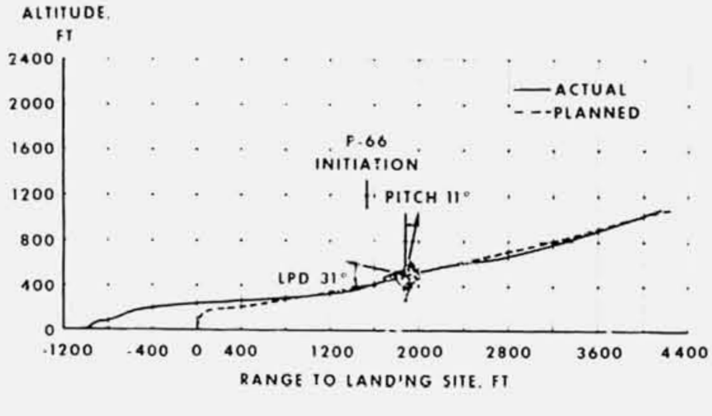


Figure 22. - Apollo 11 landing phase.

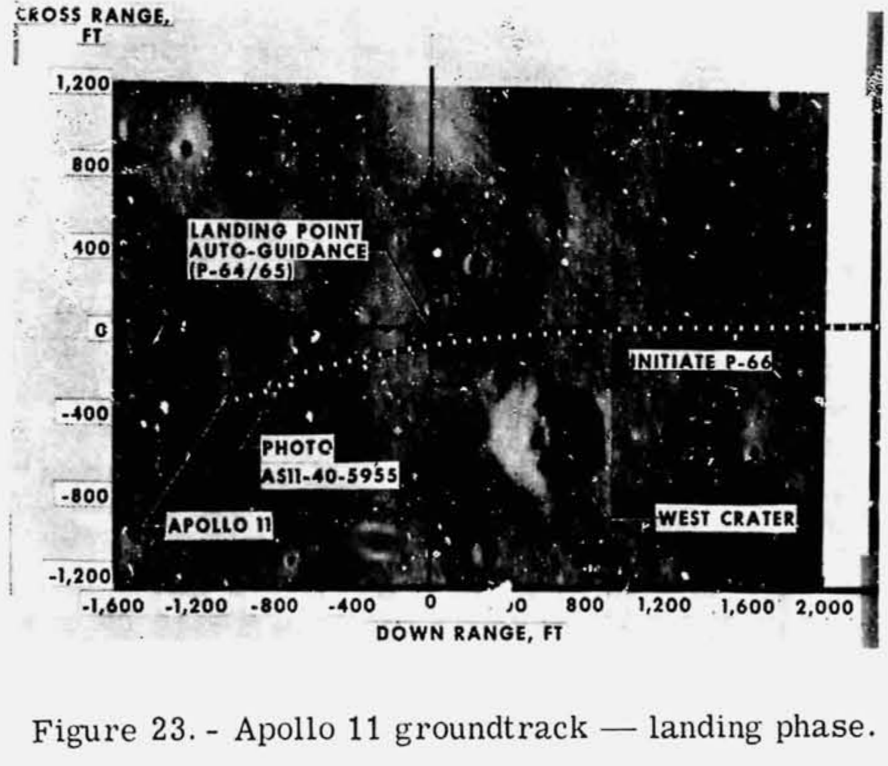


Figure 23. - Apollo 11 groundtrack — landing phase.

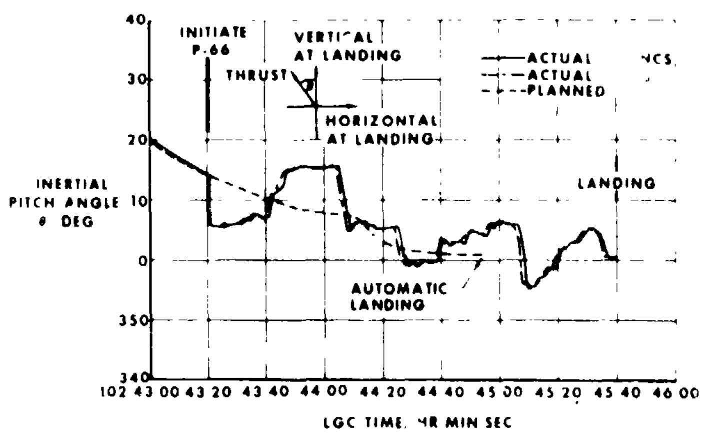


Figure 24. - Attitude profile — landing phase.

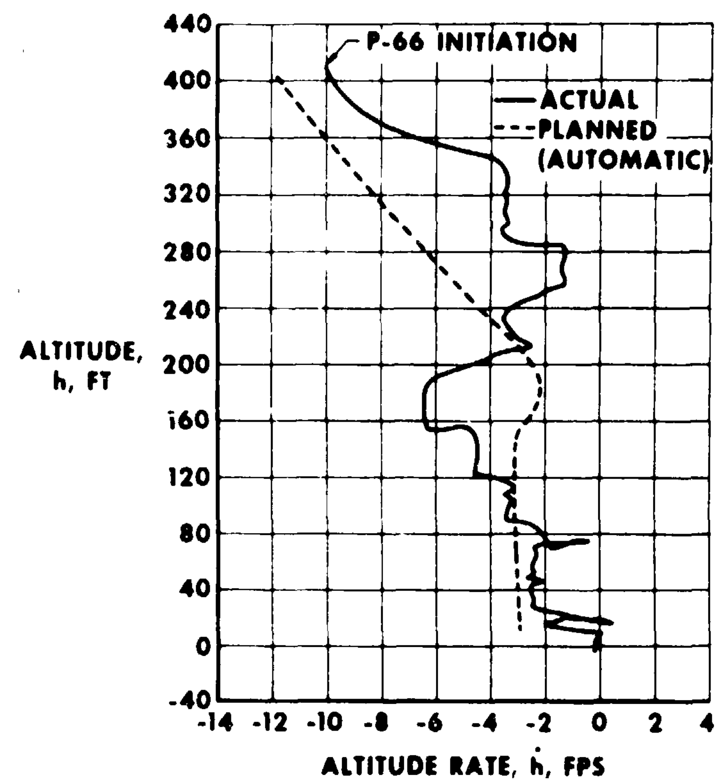


Figure 25. - Altitude/altitude-rate profile — landing phase.

The propellant situation during the landing phase is summarized in figure 26. Touchdown is shown to have occurred 40 to 50 seconds prior to propellant depletion, only 20 to 30 seconds from the landing/abort decision point, and approximately 52 to 62 seconds longer than predicted for an automatic landing. The flying time below 500 feet was approximately 2 minutes 28 seconds.

Apollo 11, the first manned lunar landing, was an unqualified success. The descent was nominal until the beginning of the landing phase (an altitude of approximately 410 feet), at which time the commander (with manual control) was required to avoid a large area of rough terrain. The size of the area was such that the crew should have been able to detect and efficiently avoid it during the approach phase, if sufficient attention could have been devoted to visual assessment. Adequate visual assessment was not possible during Apollo 11 because of the guidance program alarms. The problem causing these alarms has been corrected.

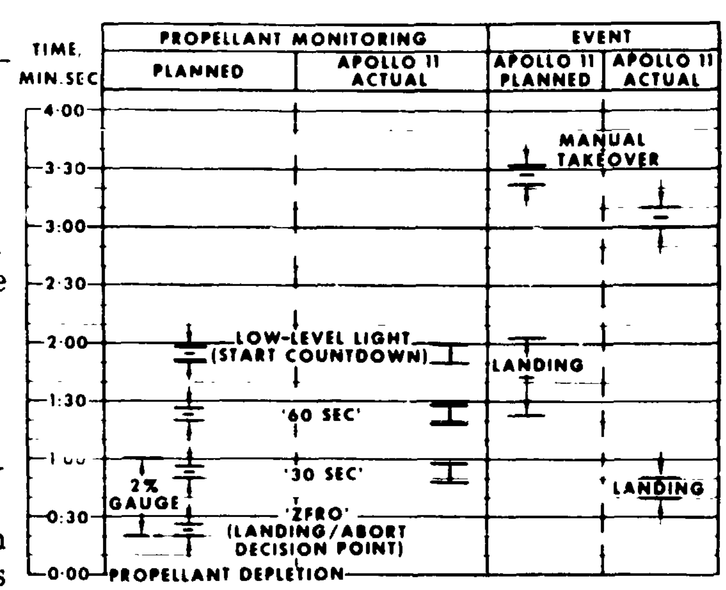


Figure 26. - Landing phase events.

Ascent. - A summary of ascent is given in table V and compared with premission estimates. In summary, this comparison indicates that no anomalies occurred during the ascent burn and that the insertion targets were closely satisfied. The 3-second difference in burn time is attributed to a slightly higher actual thrust-to-weight ratio than predicted. There is no means for determining whether the difference was due to high thrust or less weight. Usable APS propellant at cut-off was estimated to be approximately 250 pounds.

TABLE V. - APOLLO 11 ASCENT SUMMARY

(a) Events

Event	TFI, min:sec	
	Premission	Actual
End of vertical rise	0:10	0:10
Insertion (APS cut-off)	7:18	7:15
Beginning of velocity residual trim	--	7:33
Residual trim complete	--	8:37

(b) Insertion conditions

Measurement type	Altitude, ft	Radial velocity, fps	Down-range velocity, fps
Premission	60 085	32	5535.6
PGNCS (real time)	60 602	33	5537.0
AGS (real time)	60 019	30	5537.9
MSFN (real time)	61 249	35	5540.7
Postflight	60 300	32	5537.0

(c) Parameters

Ascent targets
- Radial velocity, fps 32.2
- Down-range velocity, fps 5534.9
- Cross range to be steered out, n. mi. 1.7
- Insertion altitude, ft 60 000

PGNCS velocity residuals (LM body coordinates)
- V_{gx}, fps . . . . -2.1
- V_{gy}, fps . . . . -0.1
- V_{gz}, fps 1.8

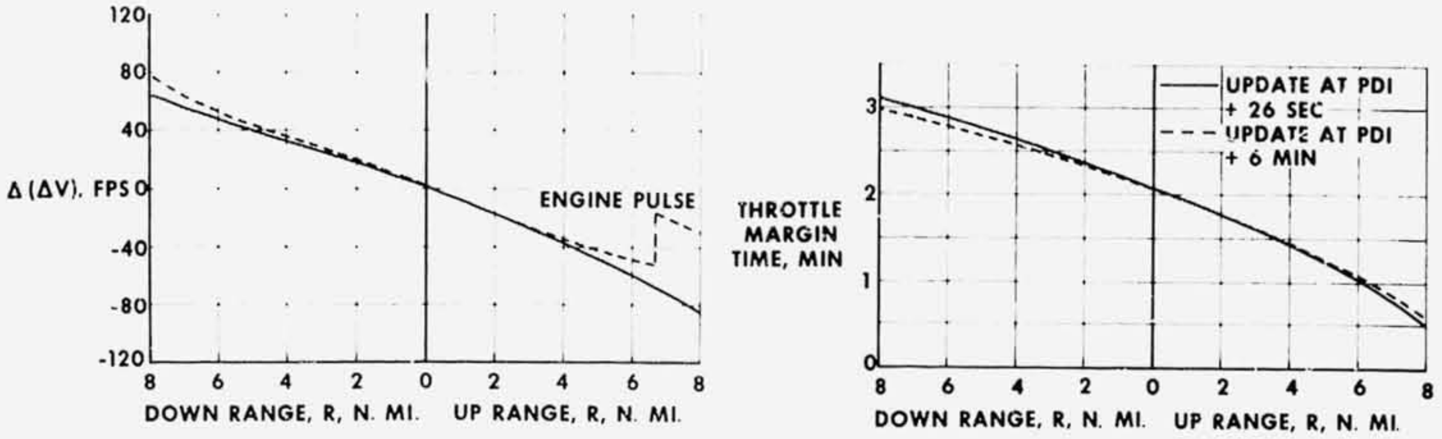
Resulting orbit after residual trim
- Apolune altitude, n. mi. 47.3
- Perilune altitude, n. mi. 9.5

Apollo 12 Planning

Apollo 12 had the same major mission objective as Apollo 11, namely, to land men on the moon and return them safely to earth. In addition, a secondary objective for Apollo 12 was to demonstrate pinpoint landing capability, required for future scientific missions, by landing within a 1.0-kilometer (0.54 nautical mile) radius of the target, near the Surveyor III spacecraft located at Apollo site 7 (latitude 3.0° N, longitude 23.4° W). Basically, the planning philosophy for Apollo 12 descent and ascent remained the same as the philosophy for Apollo 11. However, since Apollo 11 landed approximately 3 nautical miles off target and consumed more propellant for terrain avoidance than anticipated, several minor changes were considered for Apollo 12 descent. These changes were concerned with alleviating ΔV and propellant requirements and with more efficiently correcting position errors during the descent.

Two methods for alleviating propellant requirements were proposed. The first method was to perform DOI with the CSM before undocking the LM, perhaps even combining DOI with the lunar orbit insertion maneuver. By using this method, the LM ΔV and propellant requirements can be reduced by 75 fps and 190 pounds of propellant, which increases hover or translation time available in the landing phase by 20 seconds. The planning time for analysis and the crew activity time line did not permit incorporation of this method for Apollo 12. However, the method has been determined to be feasible and is currently planned for use on Apollo 13 and subsequent missions. The second method was to modulate the DPS thrust 10 to 12 times between FTP (maximum) and . percent (upper throttle region) in order to correct thrust dispersions. In using this method, the 2-minute throttle recovery region prior to high gate can be eliminated, resulting in about the same savings as the first method. This modulation requires a change to the basic guidance logic, considerable systems dispersion analysis, and DPS testing over this duty cycle before incorporating the logic. The second method also could not be incorporated in Apollo 12 planning, but is being considered for future missions. Thus, the Apollo 12 ΔV and propellant requirements for descent remained the same as the Apollo 11 ΔV and propellant requirements.

Two means for providing more efficiency in correcting position during descent were proposed. The first means was to take advantage of the detection of down-range position error by the powered flight processor during the braking phase. (See the section entitled "The PGNCS Monitoring.") Analysis showed that large updates in down-range or up-range target position could be made for small changes in ΔV and throttle recovery time (fig. 27). In addition, dispersion analysis using this update indicated that down-range dispersions would be reduced to approximately ± 1.3 nautical miles as shown in figure 28. A minor change to the guidance logic to allow the crew to manually input (through the DSKY) updates to the landing-site coordinates sent from the ground was required. The guidance change was made, and this proposed technique was approved for use on Apollo 12. The second method proposed was to change the guidance targeting for the approach and landing phases (P-64 guidance) in order to enhance redesignation (LPD) and manual maneuvering capabilities. Use of these capabilities would be required in order to reduce the 3σ dispersions shown in figure 28 to a 1-kilometer radius for pinpoint landing. The results of a limited study for varying horizontal and vertical velocity at low gate (500 feet) with vertical descent targeted to a 100-foot altitude are shown in figure 29. It was determined that by increasing forward velocity at 500 feet from 60 to 80 fps, significant gains in redesignation capability (fig. 30) were achieved while altitude rate was maintained at 16 fps. In addition,



(a) Throttle margin time.

(b) Change in characteristic velocity.

Figure 27. - Landing-site update capability during braking phase.

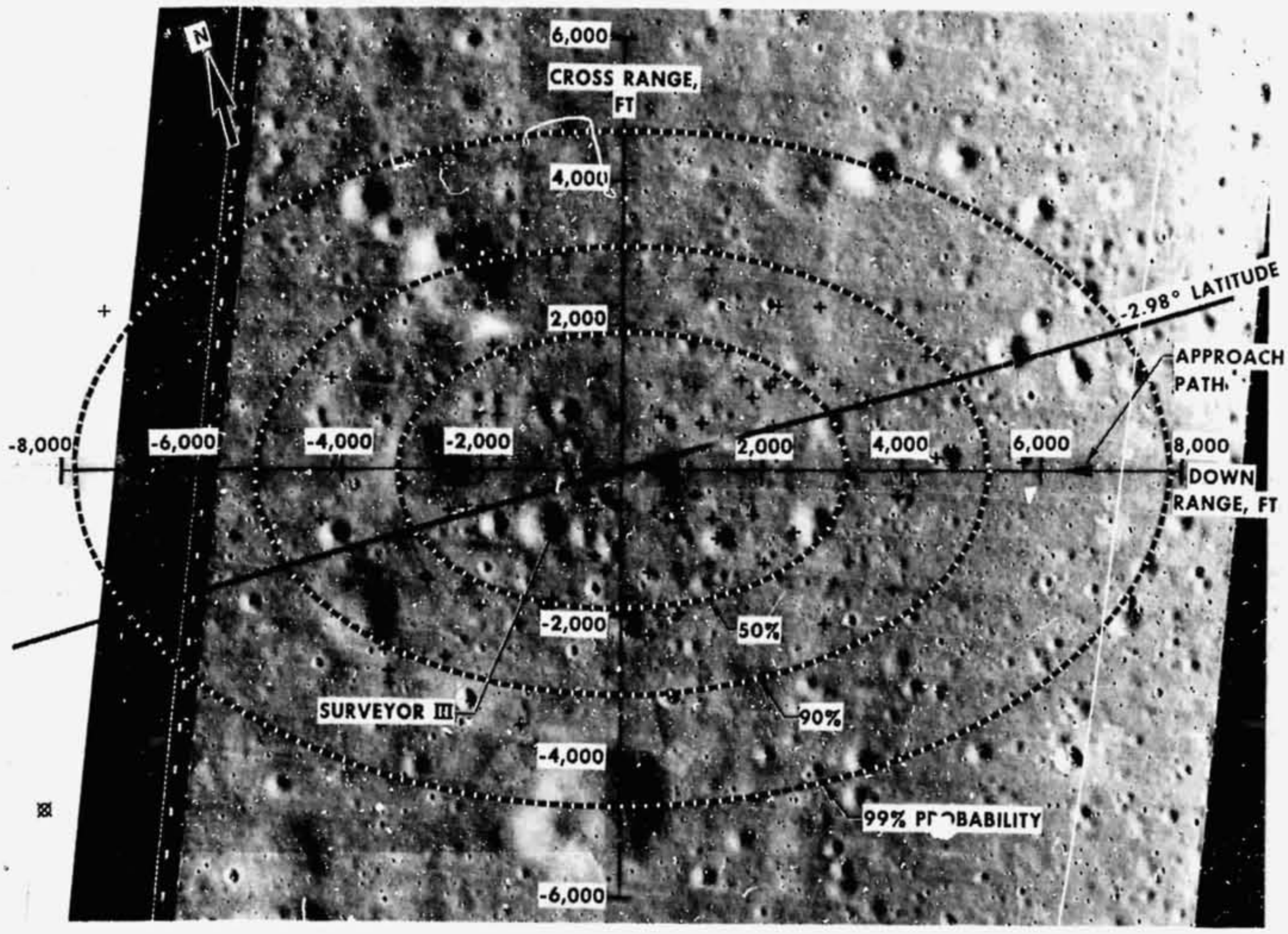


Figure 28. - Predicted Apollo 12 landing dispersions.

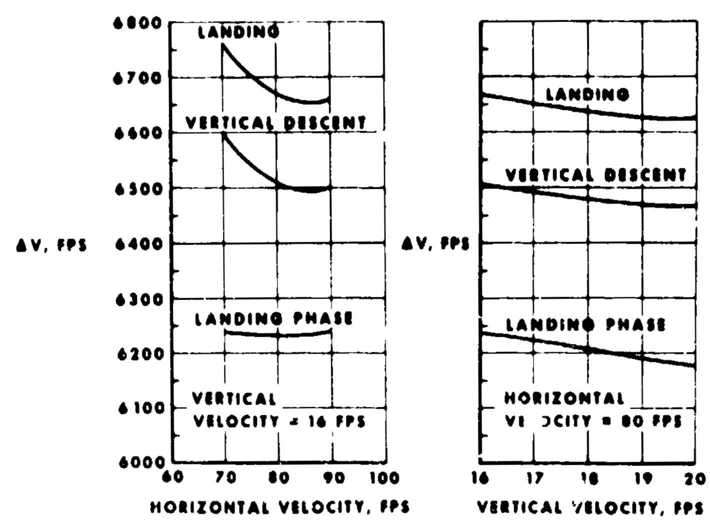


Figure 29. - Variation of ΔV with landing phase velocities.

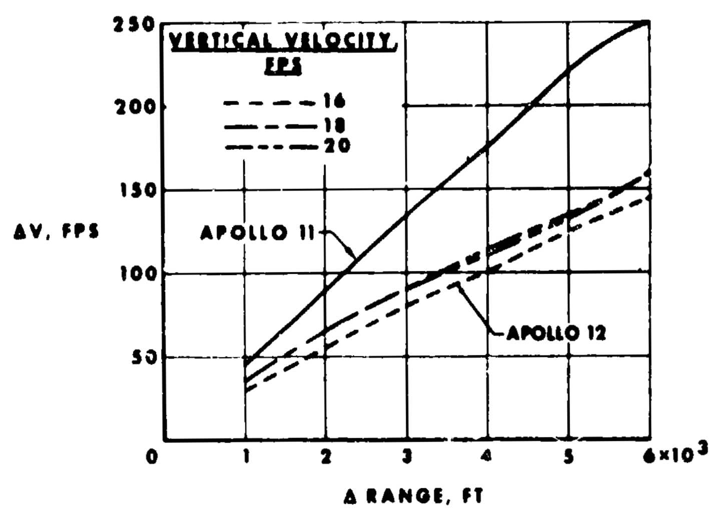


Figure 30. - The ΔV requirements for down-range redesignations at 4000-foot altitude.

this trajectory resulted in a slowly changing or more constant LPD time history during approach, as shown in figure 31. Therefore, this proposal was also accepted for the Apollo 12 operational trajectory planning.

In summary, the Apollo 12 descent and ascent used the same design as the Apollo 11 descent and ascent. The descent approach and landing phase trajectory were speeded up slightly. The capability to update the landing-site position during the braking phase was added. Finally reduction in the descent ΔV and propellant requirements for missions subsequent to Apollo 12 is contemplated.

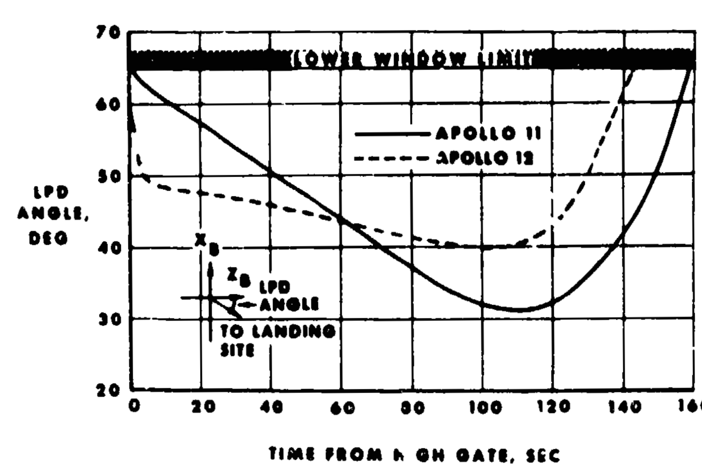
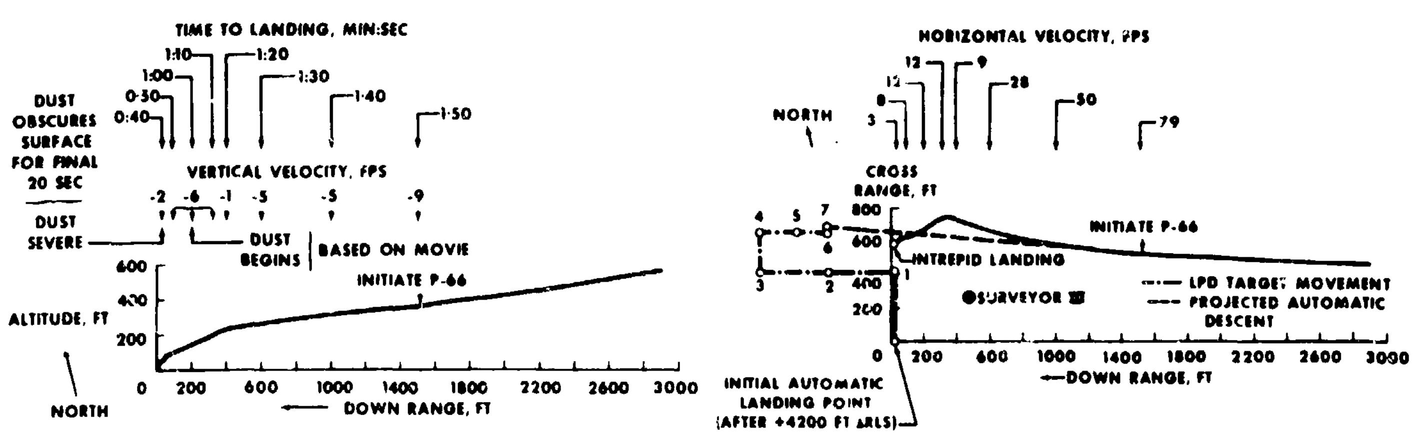


Figure 31. - Comparison of LPD profiles.

Preliminary Apollo 12 Postflight Analysis

The second manned lunar landing occurred on November 19, 1969, at Apollo site 7 in the Ocean of Storms (latitude 3.0° S, longitude 23.4° W), adjacent to the crater containing Surveyor III. As of this writing, the postflight analysis is not completed; however, a few events during the descent are worthy of comment. (The data presented in this section, since they are preliminary, are subject to change as more postflight data become available.)

During powered descent, all systems performed excellently (with not even a program alarm). The PDI occurred 5 nautical miles north of the nominal groundtrack. This cross-range distance was known to the guidance and was steered out during the braking phase for a minimal ΔV of approximately 10 fps. Also, at PDI, an up-range position error of 4200 feet was determined by the powered flight processor. Thus, the landing-site position was updated (moved down range) early in the braking phase by that amount. This resulted in a 5-second-early throttle recovery and a slight ΔV penalty (fig. 27). A down-range redesignation of 4200 feet in the approach phase could have been performed, if necessary — however, not as cheaply as the braking phase update (figs. 27 and 30). During the approach phase, the commander performed several redesignations; however, the largest is estimated to be only 800 feet. A plot of the guidance-targeted landing site as a result of these redesignations is shown in figures 32 and 33, along with a groundtrack of the landing phase trajectory under P-66 control. The time of flight in the landing phase below 500 feet is estimated to be 2 minutes, and total powered descent took 12 minutes 26 seconds (premission automatic nominal landing, 11 minutes 20 seconds). Touchdown occurred 35 seconds after low-level light ON, or approximately 60 seconds prior to the landing/abort decision point. This margin is almost twice the Apollo 11 margin. Apollo 12 stirred up more dust than Apollo 11 during final touchdown, resulting in considerable loss of visibility. What effect, if any, this will have on future mission planning has not yet been determined.



(a) Altitude history.

(b) Groundtrack.

Figure 32. - Apollo 12 groundtrack and altitude history.

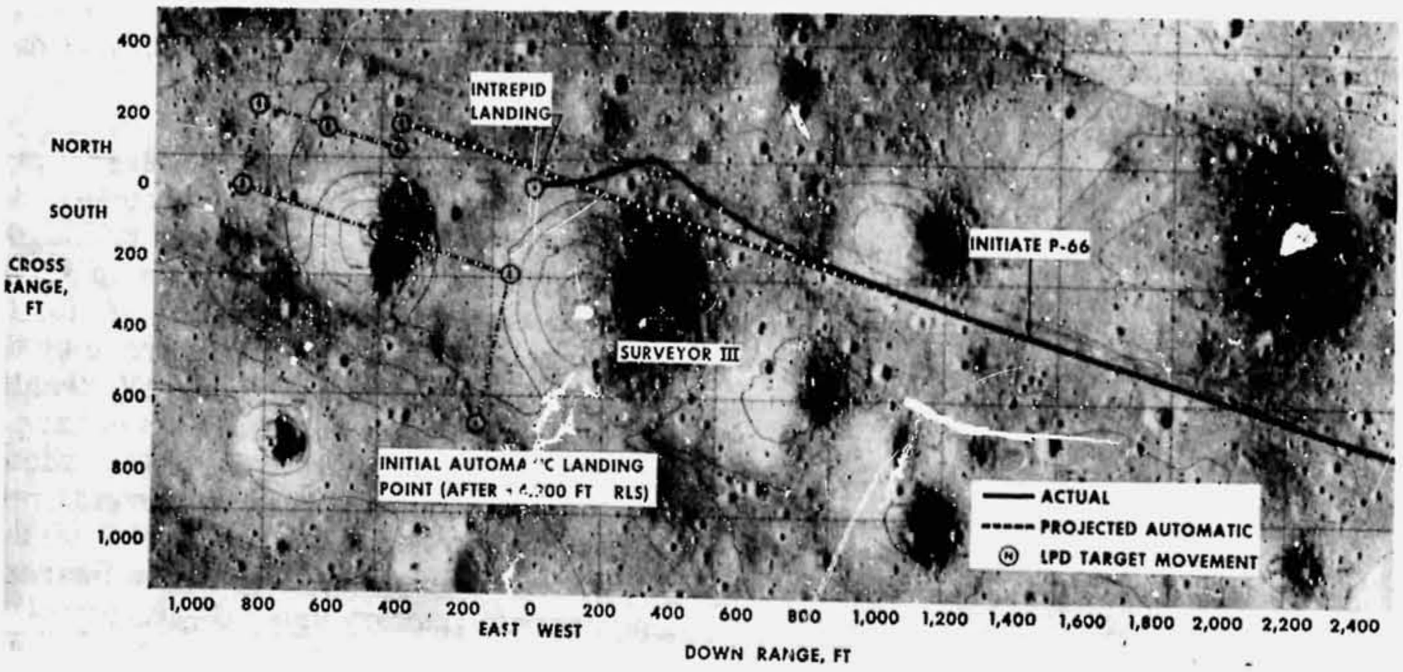


Figure 33. - Apollo 12 groundtrack — landing phase.

In summary, Apollo 12, the second highly successful manned lunar landing, achieved the first pinpoint landing. This achievement greatly enhances the possibilities for lunar exploration into the rougher mountainous areas of particular interest to the scientists.

CONCLUDING REMARKS

The premission planning for the lunar descent and ascent mission phases which led to the first, highly successful manned landing on the moon and return from the moon has been presented and compared with actual flight results. The Apollo 11 lunar module descent and ascent, the maneuvers that could be flight simulated only by actually performing the lunar landing, compared excellently with premission planning. An initial navigation error caused the landing to be approximately 3 nautical miles down range from the target, but the landing was still within the premission mapped area. The original three-phase descent design and contingency planning afforded the crew the opportunity, late in the descent, to maneuver out of an area of rough terrain to a successful touchdown.

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As a result of Apollo 11 postflight analysis, only two minor changes were incorporated in descent planning for Apollo 12. The first change was the provision of a navigation update of the landing site early in the braking phase in order to enhance pinpoint landing capability. The second change was a slight modification to the descent targeting in order to enhance the landing-site redesignation and manual translation capability in the approach and landing phases.

Apollo 12, the second highly successful manned lunar landing mission, again demonstrated excellent comparison with premission planning for descent and ascent. During descent, the landing-site navigation update and redesignation capabilities were used, along with manual maneuvering, to achieve the first pinpoint landing. The pinpoint landing, within 600 feet of the Surveyor III spacecraft, has provided confidence for premission planning of future manned lunar exploration missions.

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Houston, Texas, March 19, 1970
076-00-00-00-72

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